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HEAVY LIFT HELICOPTER FLIGHT CONTROL SYSTEM

Volume II - Primary Flight Control System Development and Feasibility Demonstration

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Due to the termination of the HLH program, reports summarizing the strides made in many of the supporting technology programs were never published. In an effort to make as much of this information available as possible, selected draft reports prepared under contract prior to termination have been edited and converted to the DOD format by the Applied Technology Laboratory. The reader will find many instances of poor legibility in drawings and charts which could not, due to the funding and manpower constraints, be redone. It is felt, however, that some benefit will be derived from their inclusion and that where essential details are missing, sufficient information exists to allow the direction of specific questions to the contractor and/or the U.S. Army.

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The U.S. Army's Heavy Lift Helicopter Advanced Technology Component Program required the design, development and in-flight feasibility demonstration of an electrohydraulic flight control system, referred to as "fly-by-wire".

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Total flight testing encompassed evaluation and demonstration in two parts. First, a direct electrical linkage was successfully demonstrated in September 1973. Subsequently, an automatic flight control system was installed and evaluated with the direct electrical linkage.

Total flight time accumulated on fly-by-wire was 315 hours. There were no inflight failures. Pilot acceptance of fly-by-wire was readily attained. Included in the program were demonstrations to military, government and industry personnel in which 163 individuals piloted the test vehicle. Feasibility of fly-by-wire flight control was clearly demonstrated.

This volume describes the primary flight control system; its design features, installation test. Included also is a report of a related activity to design and evaluate, in laboratory tests, a compatible cockpit controller subsystem in which pilot commands are converted to electrical signals. Standard pilot interface features, such as magnetic brakes, stick position trim and force trim are included. To interface with the automatic flight control system, controller driver actuators are included.

PREFACE

The subject fly-by-wire (FBW) flight control system was developed as part of the Heavy Lift Helicopter (HLH), Advanced Technology Component (ATC) Program by the Boeing Vertol Company under contract to the U. S. Army Aviation Systems Command (now the U. S. Army Aviation R&D Command), Contract Number DAAJ01-71-C-0840(P6A). The objectives of the program were: to formulate the analytical design of a fly-by-wire flight control system satisfying the HLH mission requirements, including handling qualities, control laws, and selectable modes; to design and develop critical elements; to demonstrate feasibility in flight test; and to recommend an HLH production flight control system based on the total experience.

The major elements of the fly-by-wire ATC Program and the volumes in which they are reported are listed below.

Production HLH Recommendations	Volume I
Primary Flight Control System	Volume II
Cockpit Control System	Volume II
Automatic Flight Control System	Volume III
Crewman's Load Controller	Volume III
Precision Hover Sensor	Volume III

The principal subcontractors were: the General Electric Company, Aircraft Equipment Division, Binghamton, New York; Bertea Corporation, Irvine, California; Honeywell, Inc., Minneapolis, Minnesota; and RCA, Camden, New Jersey.

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INTRODUCTION

Fly-by-wire (FBW) primary flight control systems (PFCS) use electrical signaling to provide the desired responses to pilot commands without a mechanical connection between the cockpit controls and the swashplate actuators. In the HLH system, no aircraft motion sensors are incorporated, as is frequently the case for fixed-wing aircraft. The primary flight control system is analogous to unaugmented mechanical control. Stability and control augmentation functions are derived in an Automatic Flight Control System (AFCS). This particular flight control system configuration was the result of a FCS concept selection which was approved by the Army in January, 1972. The selected concept is illustrated in the block diagram of Figure 1. Feasibility demonstration flight testing used the Boeing Model 347 helicopter shown in Figure 2. Flight tests of the PFCS were successfully completed in September, 1973, after 18 hours and 49 minutes of flight. Flight tests of the AFCS were completed in October, 1974. Total flight time was 315 hours, including demonstration flights.

Part I of this volume summarizes the DELS FBW demonstration system. Part II summarizes the development and testing of the HLH cockpit control subsystem (CCS).

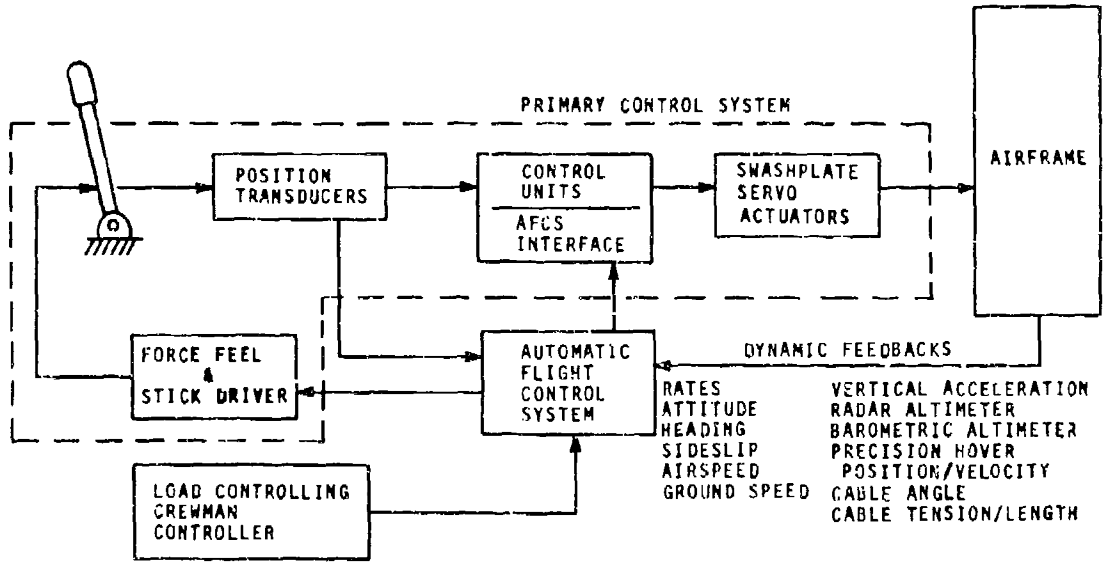


Figure 1. Selected HLH Flight Control System Configuration.

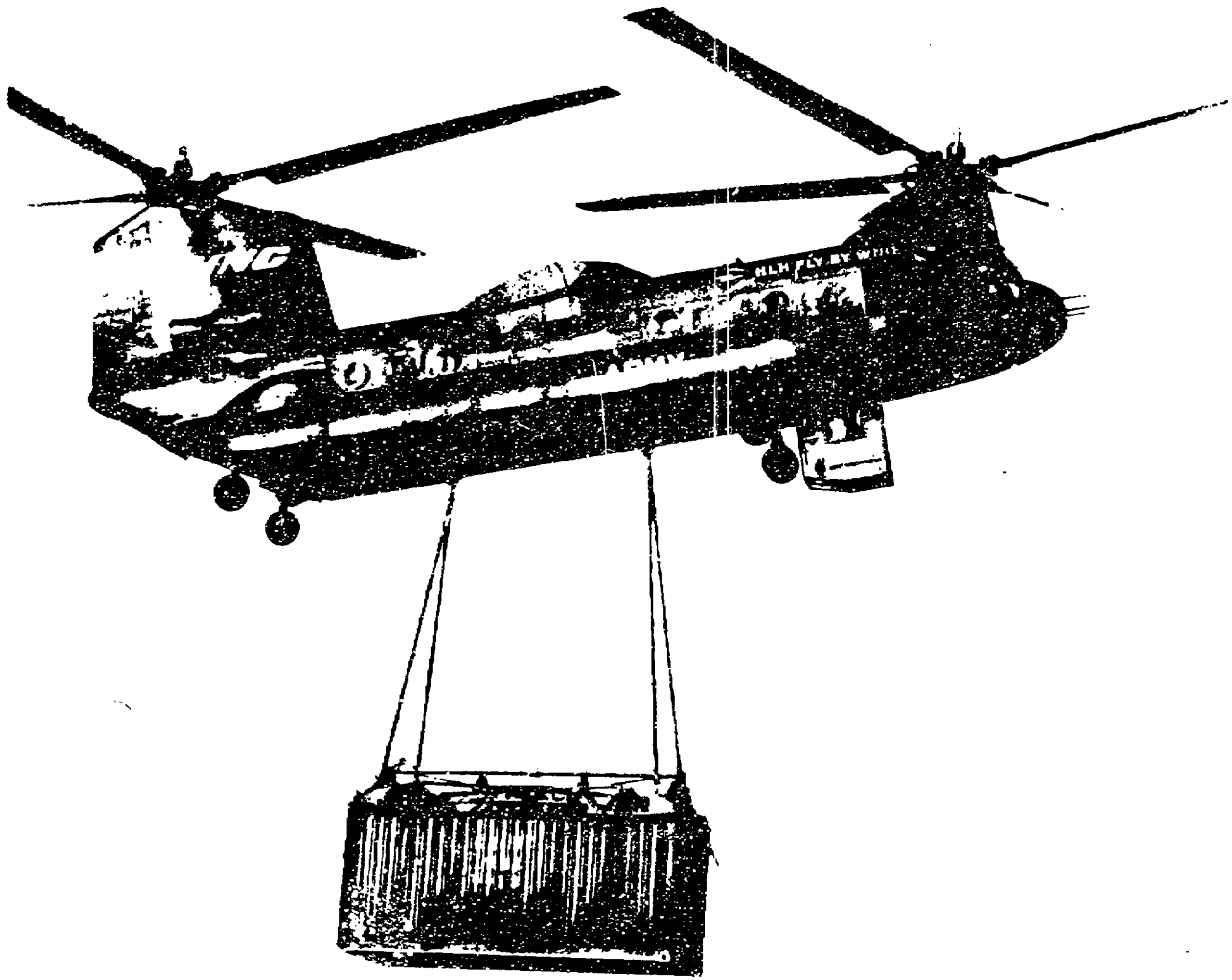


Figure 2. Fly-By-Wire Flight Test Helicopter.

PART 1 - PRIMARY FLIGHT CONTROL SYSTEM

GENERAL DESCRIPTION OF THE ADVANCED TECHNOLOGY COMPONENT (ATC) PRIMARY FLIGHT CONTROL SYSTEM

The critical elements designed and tested in the ATC Program are a direct electrical linkage system (DELS) and a cockpit control subsystem. The DELS, which had to be configured for the test vehicle, constitutes the stick position transducers, the DELS Control Units (FBW electronics), the swashplate driver actuators, and associated control/status panels.

The cockpit control subsystem consists of grips, cyclic and collective control sticks, rudder pedals, synchronization torsion linkages, transducers, and programmable force feel/cockpit controller driver actuators. This HLH subsystem was submitted to laboratory tests only. In the test vehicle, the existing CH-47 cockpit controls, force feel springs, magnetic brakes, and controller driver actuators were utilized.

Direct Electrical Linkage Subsystem (DELS)

The DELS is an electrical equivalent of the conventional, mechanical primary flight control system for a tandem rotor helicopter. The following paragraphs will identify the DELS equipment and describe its installation in the test helicopter. A block diagram of the DELS and ancillary aircraft equipment is shown in Figure 3. Redundancy level of each item is indicated. Operation of the system and its equipment is described in subsequent paragraphs.

Stick Position Transducers (SPT)

These devices sense cockpit control positions and feed corresponding electrical signals to the DELS control units. A SPT (Figure 4) contains three linear variable differential transformers (LVDT). Each control axis drives two SPT assemblies for a total of six LVDTs. This arrangement provides independent triplex signal sources of two LVDTs each.

DELS Control Unit

The DEL control units pictured in Figure 5 mix, limit, and amplify the SPT signals and feed them to the swashplate driver actuators. There is one control unit for each of the three channels. All the DELS electronic circuits are located in the

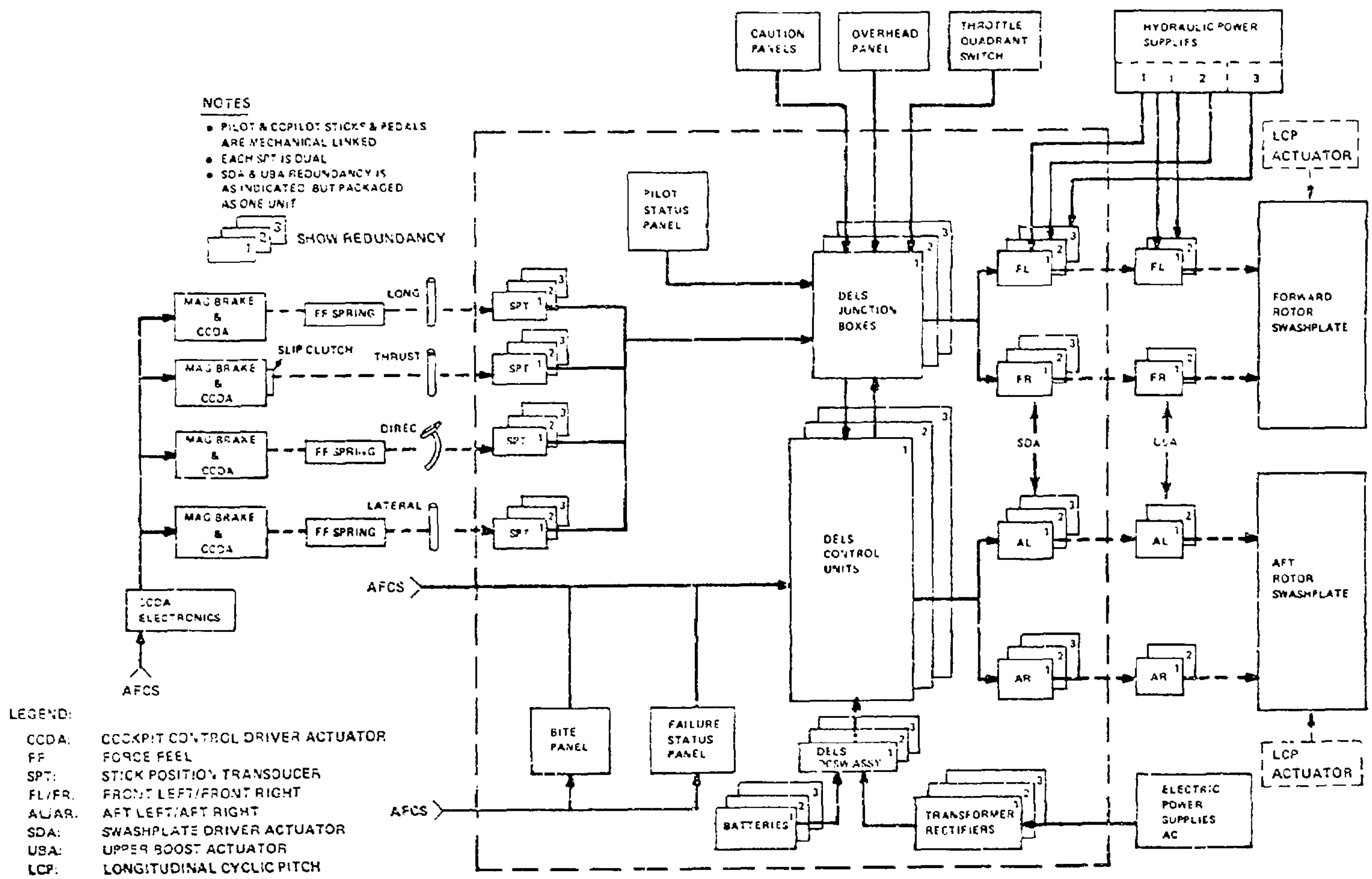


Figure 3. Direct Electrical Linkage Subsystem (DELS) Equipment Diagram.

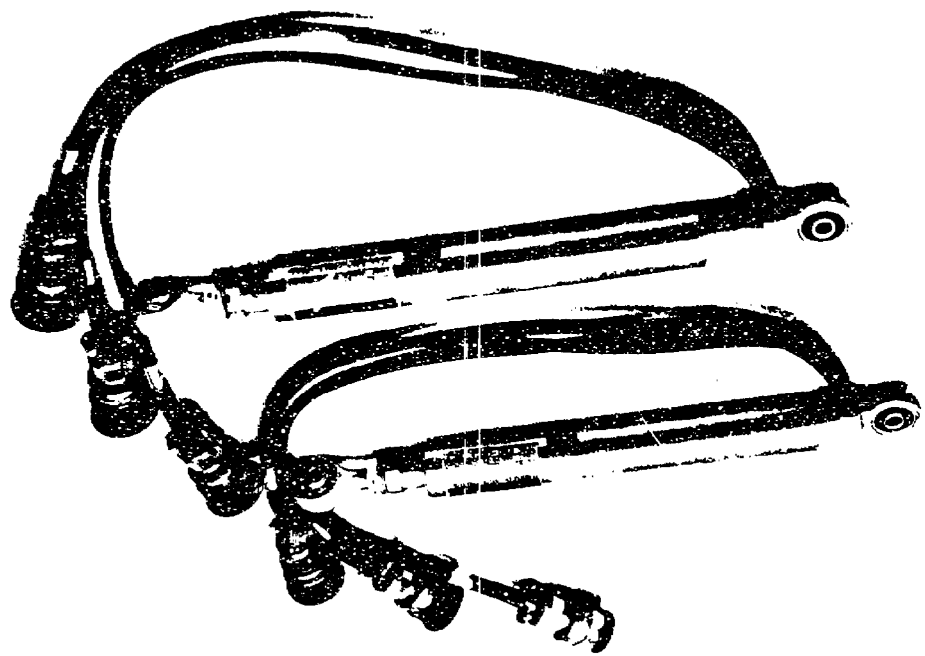


Figure 4. Stick Position Transducers.

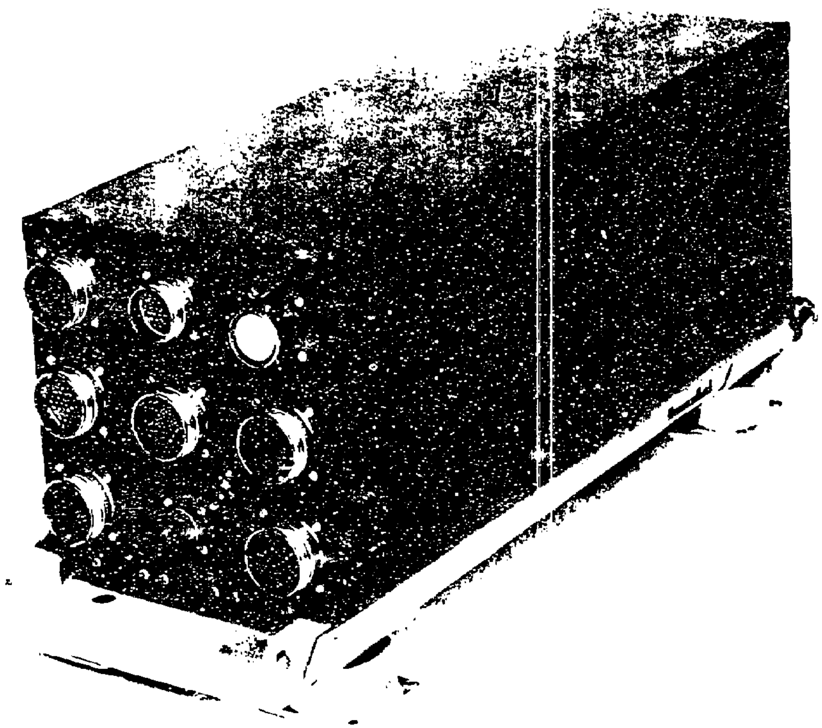


Figure 5. DELS Control Units.

control units. These include failure detection, built-in test, power conversion, and AFCS interface circuits. The function of mixing is to transform cockpit control inputs into swashplate motions.

Swashplate Driver Actuators (SDA)

The swashplate driver actuators (SDA) are part of the DELS. They drive the mechanical input mechanism on the upper boost actuators. These, in turn, control the tilt of the nonrotating swashplates at each rotor head. Each SDA (Figure 6) is a triple-channel electro-hydraulic unit. It consists of three independent conventional hydraulic actuators connected to a common output fitting.

Pilot Status Panel

The pilot status panel (Figure 7) has three light indicators to inform the pilot of a DELS channel failure. When appropriate, the pilot can reset a failed channel by depressing the indicator. The panel also has a built-in test equipment (BITE) armed switch and indicator light.

DELS Status Panel

This status panel (Figure 8) has fourteen red indicator lights and an indicator reset switch for each channel. These lights can show which portions of a channel have failed. Four green lights for each channel indicate which are the active actuators.

BITE Panel

The built-in test equipment (BITE) panel (Figure 9) is used for preflight checkout. Tests are initiated for individual channels. Once initiated, the BITE sequentially tests each failure detection circuit in the channel and lights a GO indicator for the channel if all tests were successful. When a test is unsuccessful, the sequence stops and the number of the failed test is displayed. The "box controls" test provides a check of channel tracking.

DELS Junction Boxes

Three junction boxes (one per channel) serve to minimize the number of connectors needed on the control units. They accept wire inputs from the SPTs, the ship's overhead and caution panels, and the DELS pilot status panel. They interface with their respective control units through two jacketed cables. For installation expedience on the test vehicle, they interface with the forward driver actuators.

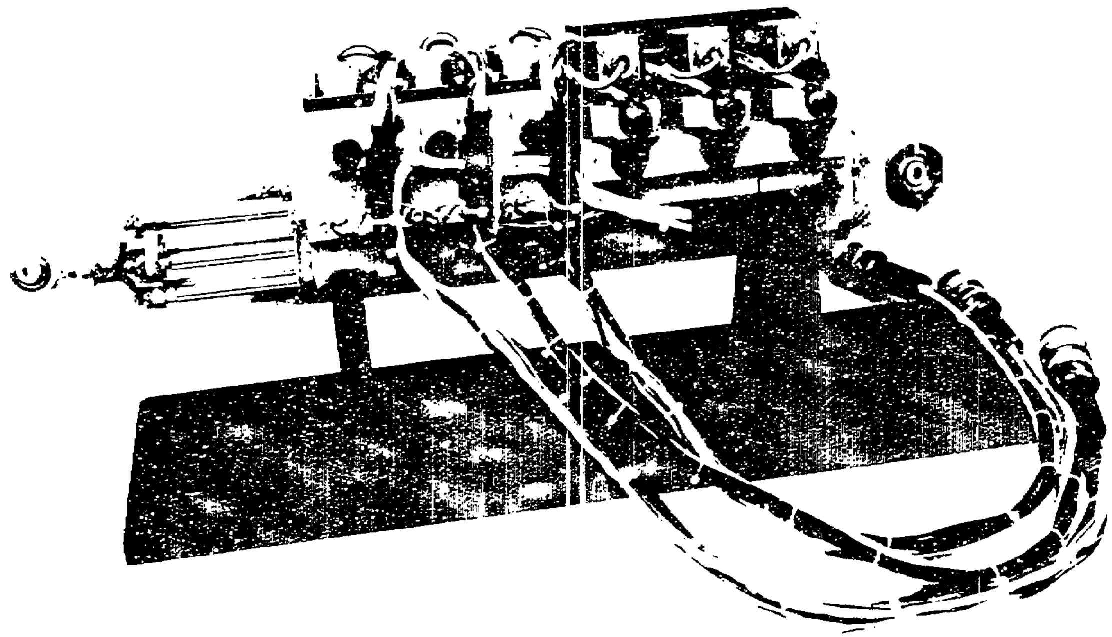


Figure 6. Swashplate Driver Actuator.

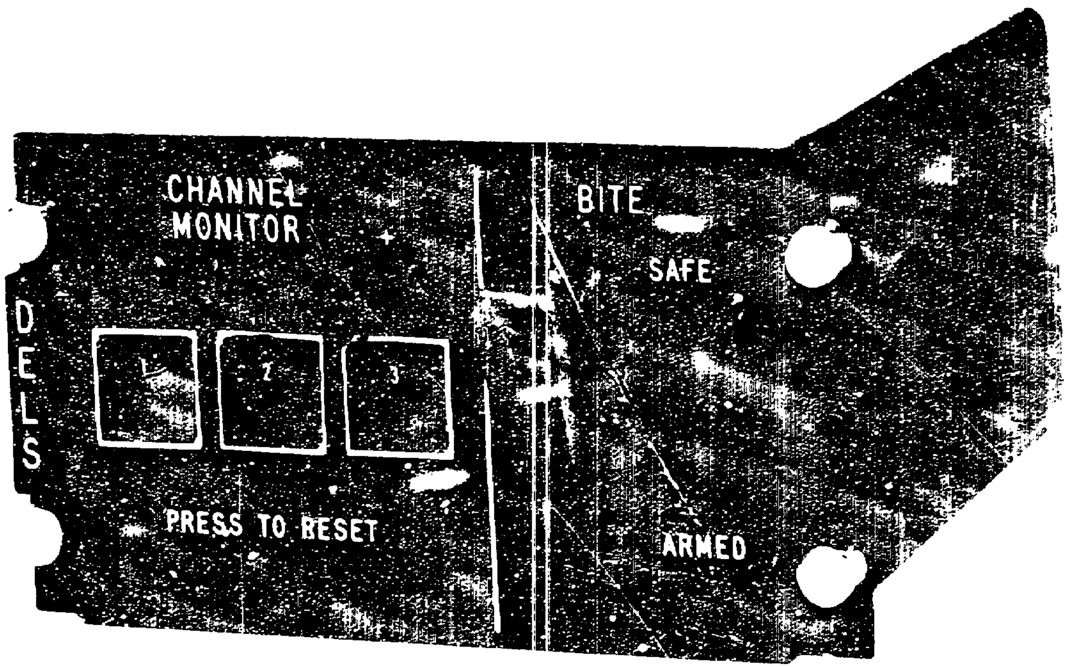


Figure 7. Pilot Status Panel.

Figure 8. DELS Status Panel

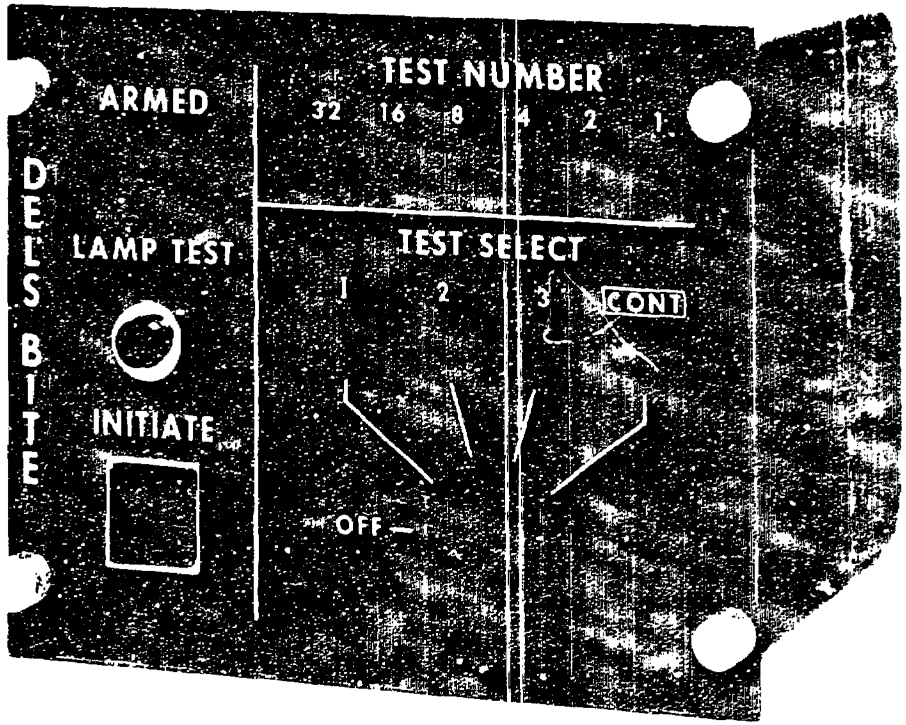


Figure 9. Built-In Test Equipment (BITE) Panel.

Electric Power Supplies

Only DC power is required for the DELS. Each DELS channel has a transformer rectifier and a backup battery to supply its basic power needs. The transformer rectifiers receive three-phase, 400-Hz 115 VAC from the aircraft's generators. The use of three DC supplies satisfied the integrity of triplicated DELS for the feasibility demonstration. A DELS DC switching assembly was incorporated to bring the backup batteries on line upon loss of all primary electrical power.

Hydraulic Power Supplies

The aircraft hydraulic system's 3000 psi was reduced to 1500 psi for two of the swashplate driver actuator channels. The aircraft's utility hydraulic system was utilized to supply the third channel in the swashplate driver actuators. Thereby, independent supplies at 1500 psi are provided for each channel.

Upper-Boost Actuators

Existing dual CH-47 upper-boost actuators were retained on the feasibility demonstration vehicle. Control was from the output ram of the triplicated swashplate driver actuators.

LCP Actuators (Speed Trim)

The third support point required to define the plane of the swashplate was provided by the existing LCP speed trim actuator. This actuator and its function were not directly involved with the DELS operations.

Other Interfaces

An interlock interface with the throttle quadrant prevents operation of BITE except when throttles are in "STOP" position. Electrical connections between the DELS and the caution panel are provided to allow indication of DELS (and AFCS) failures on the panel. The overhead panel contains two circuit breakers from which the caution panel circuits are energized.

Each DEL control unit has an AFCS interface which receives four electrical control axis signals from each of the three AFCS Input/Output Processors. In the interface, median voting is utilized to select the median signal for each axis.

DELS Installation In Test Helicopter

Equipment installation was made in a manner to minimize airframe modifications wherever the integrity of the feasibility demonstration would not be degraded. Installation locations were as shown in Figure 10.

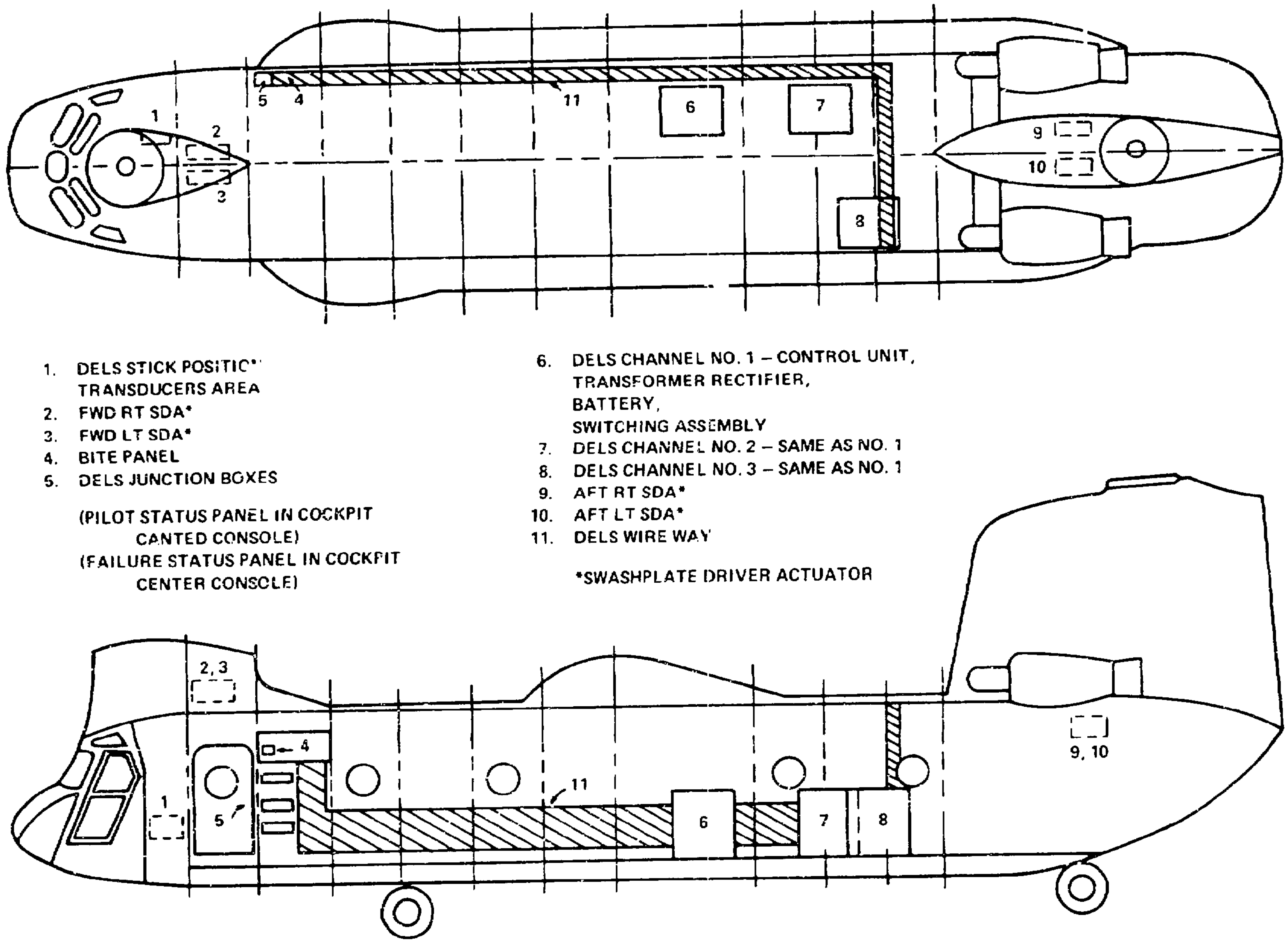


Figure 10. DELS Installation in Test Helicopter.

Figure 11 shows a typical cabin rack, location 8 in Figure 10. The channel 3 DELS control unit, transformer rectifier and battery are mounted. Later, an AFCS channel was installed in the available rack space.

The aft left driver actuator is shown in Figure 12, location 10 in Figure 10. The forward left driver actuator is shown in Figure 13, location 3 in Figure 10. The cabin heater was removed and the SPTs were installed in the heater closet; location 1 of Figure 10.

Flight Test Vehicle

The test vehicle was a modified CH-47 helicopter, S/N 65-7992. Prior to the HLH ATC program, the aircraft had been used in the Model 347 Development Program. In this program, an experimental wing had been installed approximately mid-fuselage top. Structural changes for wing attachment and rotation were made. For the ATC FBW Program, the wing was removed. However, structural changes to airframe were not removed but were faired over. This accounts for the bump on top of the aircraft. The bump had no effect on the FBW demonstration testing.

Basic changes to the 347 helicopter relative to a CH-47 are:

1. Fuselage length, 110 inches greater.

2. Aft pylon height increased thirty inches.

3. Four-bladed rotor which used standard CH-47C blades, but with trim tabs slotted into quarters.

4. Forward rotor delta 3.

5. No vibration absorbers installed.

The engines were Lycoming T55-L-11C with contingency power provisions to provide 4,600 SHP at 235 N_R under sea level/standard day conditions.

DELS Principles Of Operation

The DELS is a dual fail operative system using the triplex self-monitor active/on-line concept. It has three identical channels which link the pilot's controls to the upper boost actuators. A functional block diagram of one channel is shown in Figure 14.

Each DELS channel has an active and model channel. Comparators in the two sides shut the channel down if the two sides do not agree. When a channel shutdown occurs, the

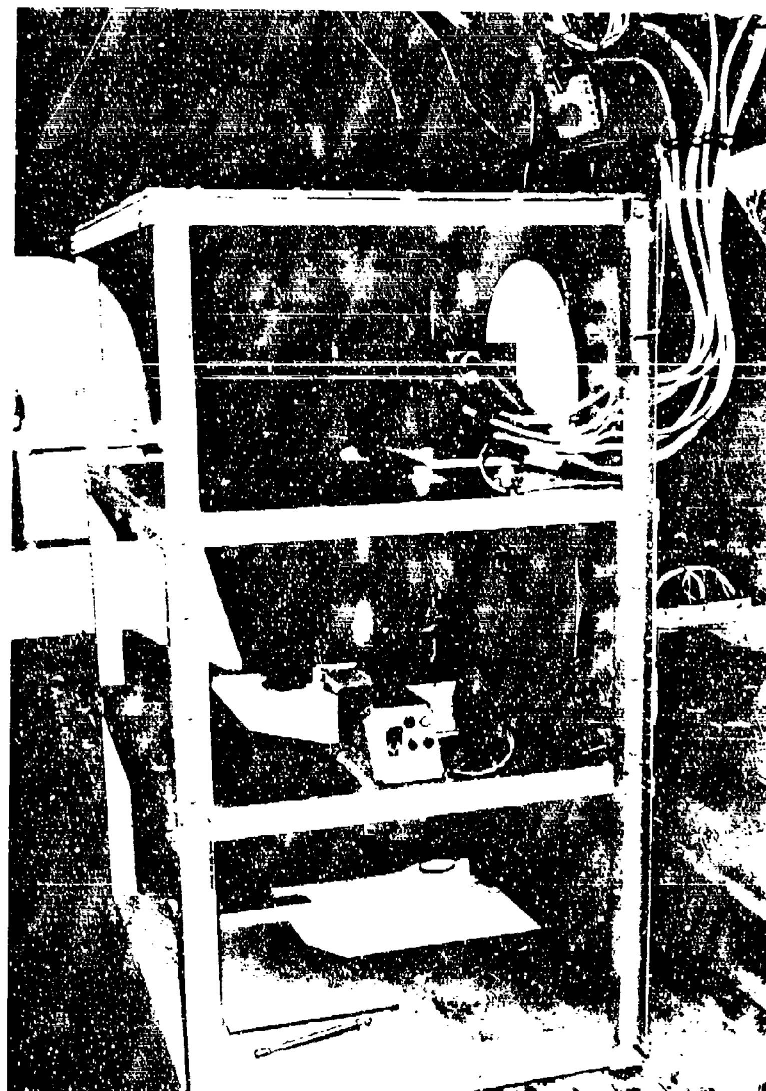


Figure 11. Cabin Equipment Rack.

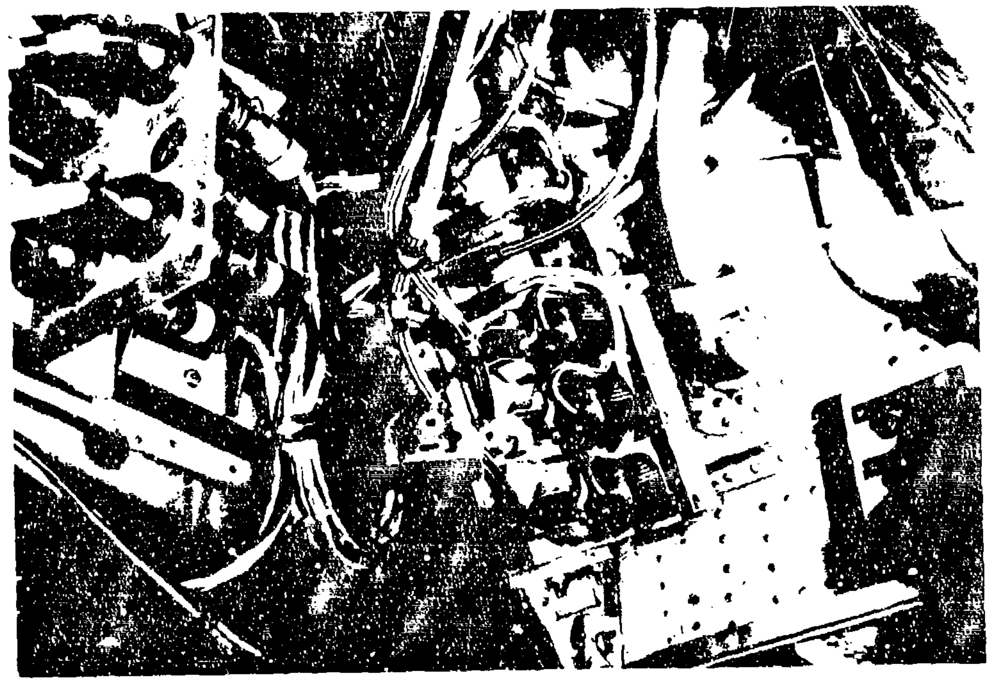


Figure 12. Left Aft Actuator Installed.

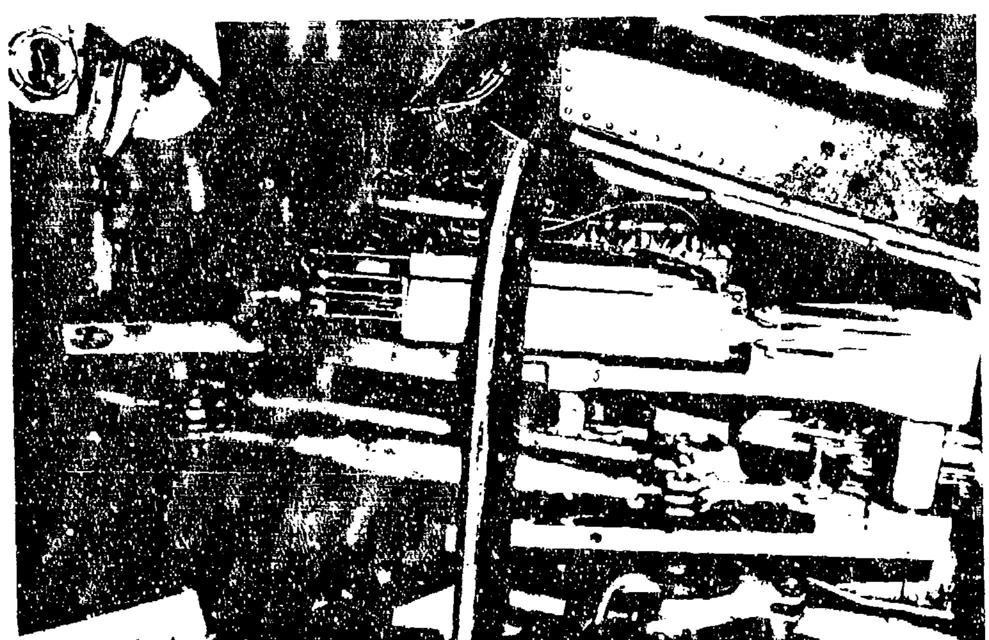


Figure 13. Forward Left Driver Actuator Installation.

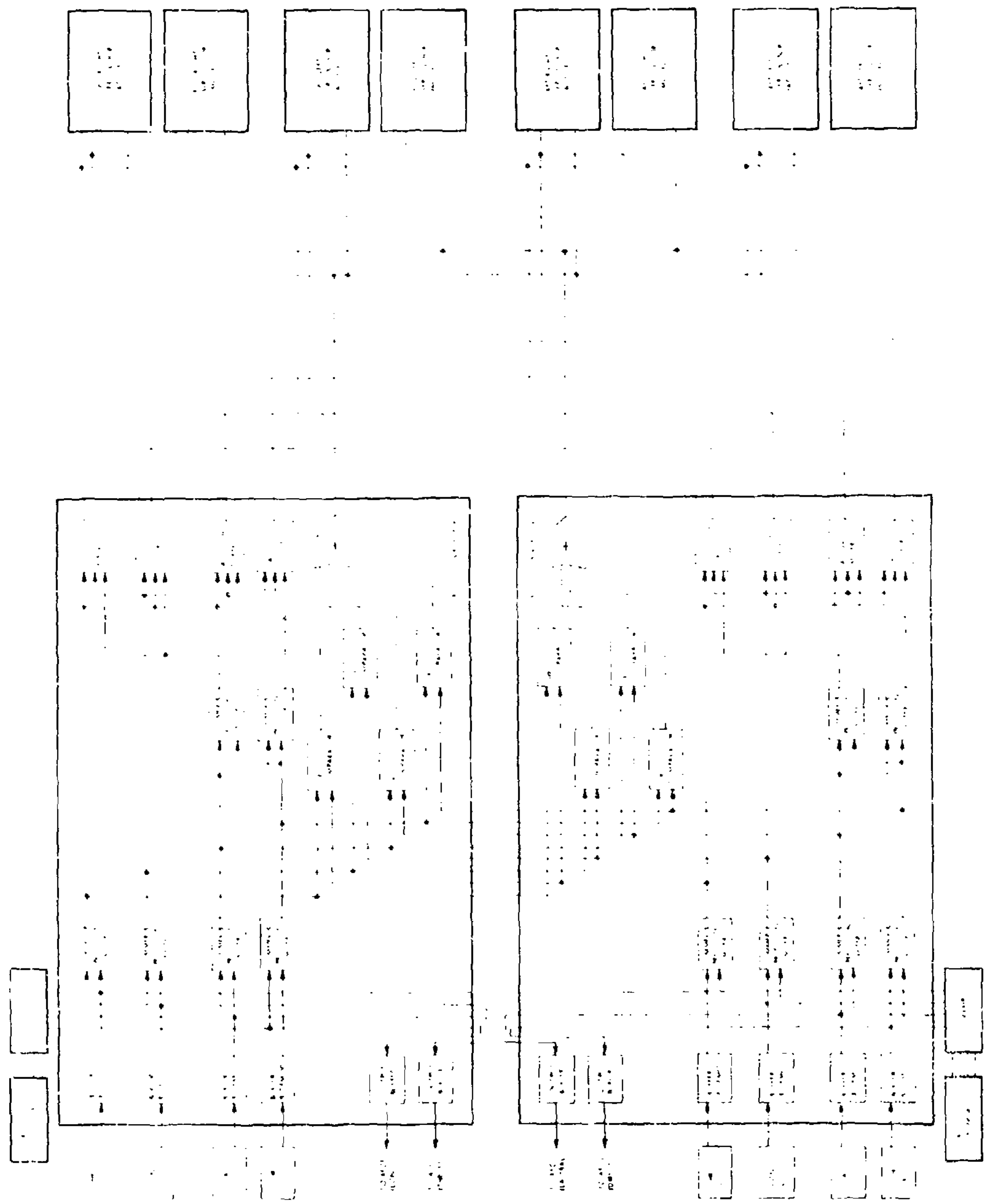


Figure 14. DELS Functional Block Diagram.

matching channel section in each of the four swashplate driver actuators is hydraulically bypassed.

When the system is operating, one channel is active and the other two are on-line. The active channel is the electrical link. The on-line channels are also operating, but do not produce an output to the swashplate driver actuators. See Servo Loop, below.

A differential pressure signal from the driver actuator causes the servo-amplifier to follow ram movements. This prevents an on-line servo-amplifier from loading the ram if there is a slight difference between channel outputs. In case of a failure, the effects of the differential pressure signal are limited by circuitry so that the on-line channel can oppose a failed active channel after an actuator displacement of about .08 inch. When the active channel is shut down because a failure has been detected, one of the on-line channels becomes active.

Control Position Sensing and Demodulation

Cockpit control positions are sensed by linear variable differential transformers (LVDT) in the stick position transducers. The LVDT output is a 1,800 Hz signal whose amplitude and phase are proportional to control position. The LVDT outputs are fed to stick mixer circuit boards in the control units. There, they are buffered and demodulated to produce DC signals with polarities and amplitudes proportional to the control positions. These signals are then applied to summing amplifiers (summer and precision limiter in Figure 14). The demodulated signals are also sent to the AFCS.

AFCS Interface

A function block diagram of the AFCS interface is shown in Figure 15. The interface consists of two identical circuit boards which receive four control axis signals from each of three AFCS input/output processors.

Each of the two boards has six input buffers. On one board, three of the buffers are for the thrust AFCS inputs. On the other board, three of the buffers are for the lateral AFCS inputs and three are for the directional AFCS inputs. The buffered signals are exchanged between boards so that each multiplexer has three inputs for each of the four control axes.

The multiplexer is a system of electronic switches that selects the control axis signals in a rotating sequence. It applies three signals per control axis to the voters at any one time. The multiplexer allows the four control axes to time share one set of voters.

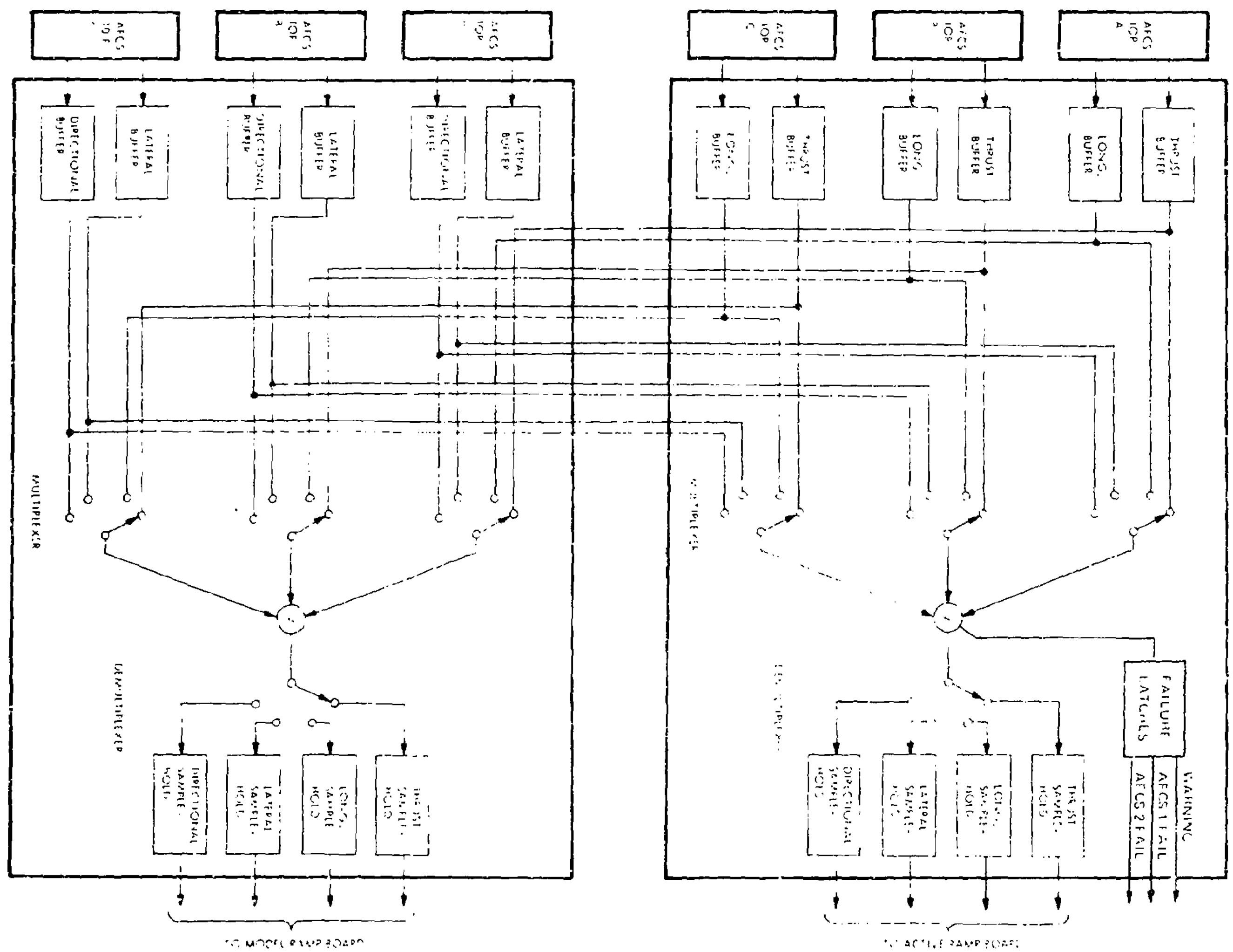


Figure 15. Automatic Flight Control System (AFCS) Interface Block Diagram.

A voter consists of switching and comparator circuits. It selects the median of its three inputs and switches it to the demultiplexer. It also provides outputs to indicate the failure status of its inputs.

The demultiplexers are synchronized with the multiplexers. They are electronic switches which feed the voter outputs to the correct sample hold circuits. The sample hold circuits are buffer amplifiers with very long time-constant networks. They hold their outputs constant during the time they are disconnected from the voter.

Ramp Function Generator

A block diagram of the ramp function generator is shown in Figure 16. Both the active and model ramp circuit boards have three of these circuits. Each AFCS input axis goes through a ramp circuit, except thrust, Figure 15. Thrust signals are only limited on the ramp board.

The DELS combines the cockpit control movements with AFCS commands to position the actuators. Important to flight safety is the "frequency splitter" function in the ramp generator. The AFCS signal is split into trim and dynamic compensation paths. The trim path, noted on Figure 16, provides long-term trim correction of a low frequency nature, such as directional pedal offset with airspeed. High frequency compensation, such as yaw rate damping, is provided by the dynamic path. Separate amplitude limits are included in each path. Cross signaling from the static path continually recenters the dynamic path.

Under steady state conditions, the output of the ramp generator is equal to its input. Therefore, the input to the high-pass limiter is zero and the input to the stick/mixer is the ramp generator output (trim value). A change in input to the ramp generator produces an immediate change in the high pass limiter output (dynamic value) and the stick/mixer input. As the generator output ramps to the new value, the high pass limiter output ramps to zero. If the input change is greater than the high pass limit, the stick/mixer input is initially equal to the high pass limit. It then slowly changes to the output of the low pass limiter.

If the voter in the AFCS interface detects a second failure, it grounds out the low pass limiter input and the high pass limiter output. Switches S_1 and S_2 in Figure 17 will be closed. The stick/mixer input drops immediately to the ramp generator output and then ramps to zero as the ramp generator responds to the grounding of its input. Both active and model ramp generator outputs will return to zero at the same rate. Therefore, the stick/mixer comparators will not detect the failure and the DELS failure status will not be affected by the AFCS

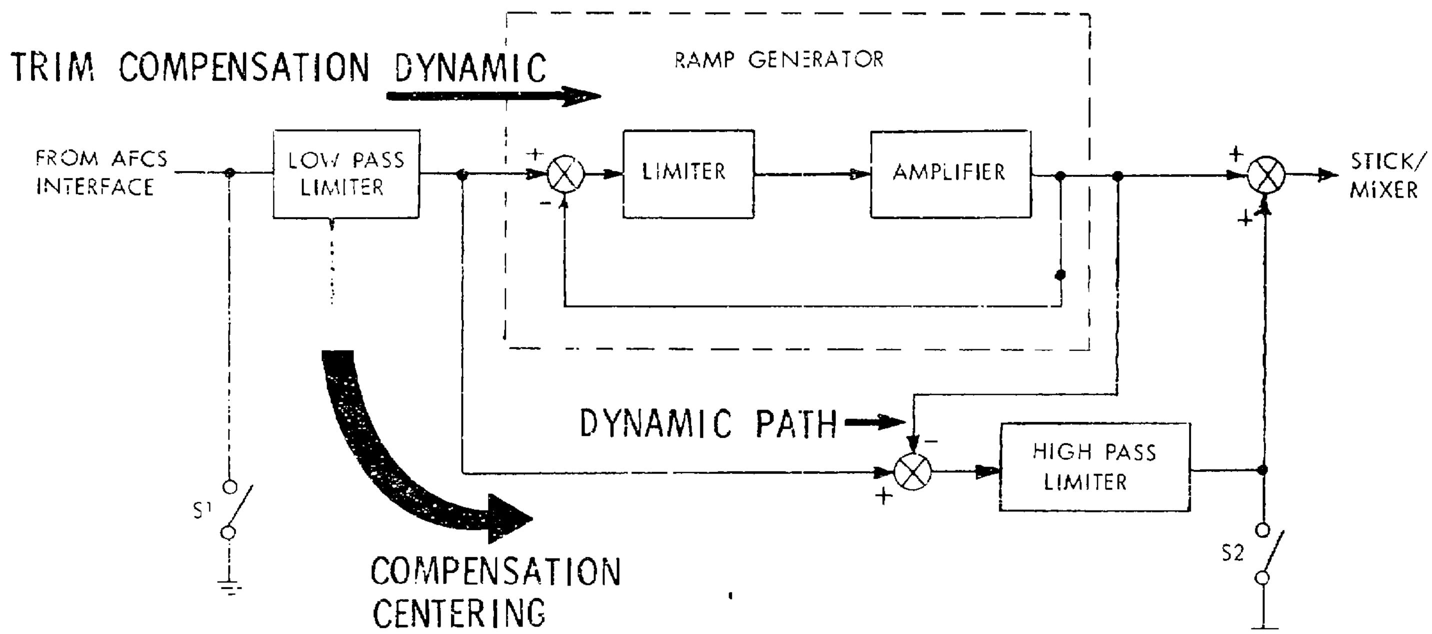


Figure 16. Ramp Function Generator Block Diagram.

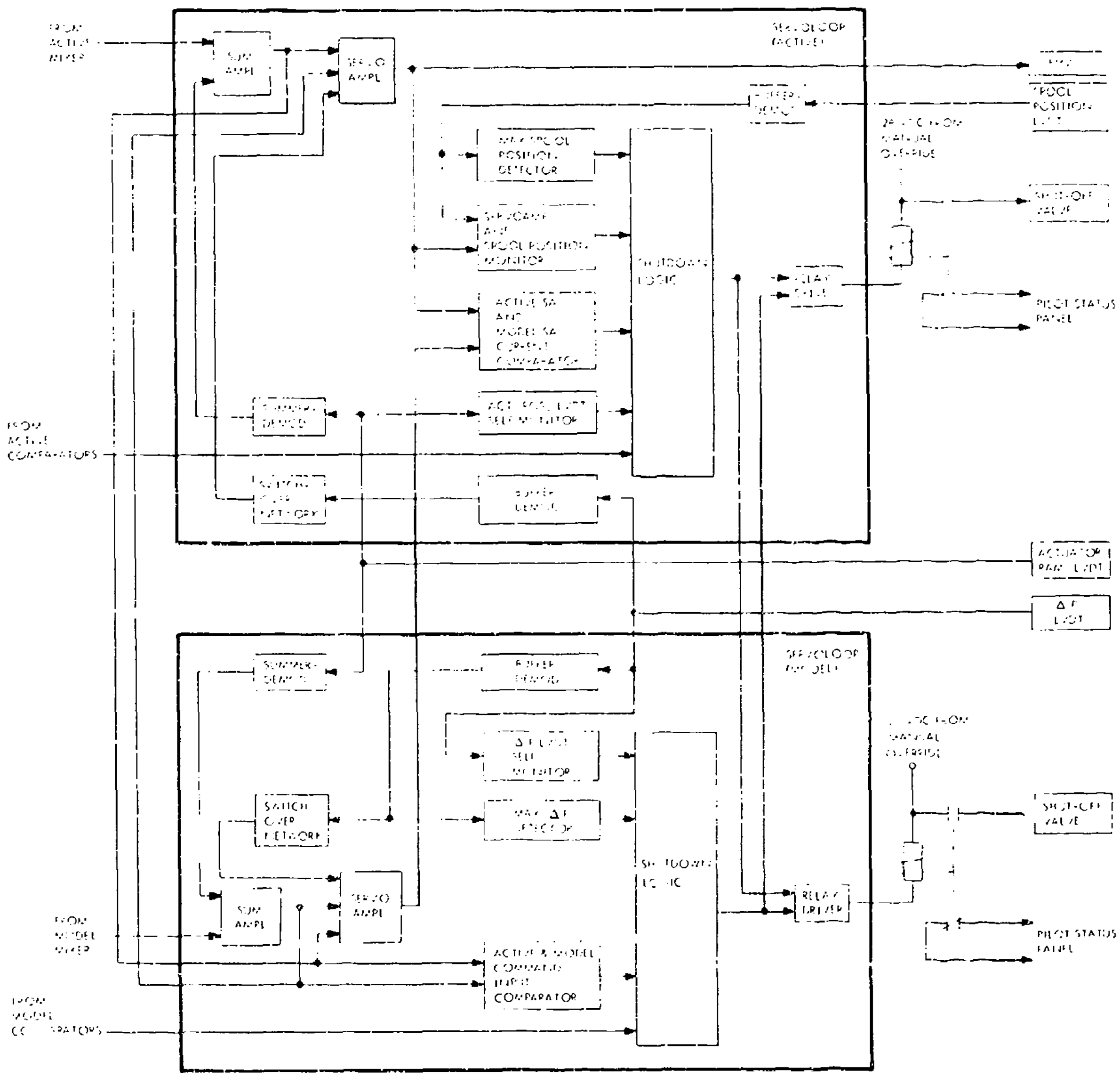


Figure 17. Servo Loop Block Diagram, Single Actuator Channel.

input failures.

Also, in the event of an AFCS hard-over (extremely remote possibility), the authority limits and the ramp generator (rate limit) were selected to keep short term impulse disturbances, and long term trim changes within safe limits. In other words, pilot recovery time upon multi-axis hard-over is adquate for flight safety.

Summing and Limiting

Each demodulated LVDT signal is summed with the corresponding AFCS signal from the ramp generator, Figure 14. The sum of the two inputs is passed through a limiter to prevent an input in one axis from utilizing more than a specified percentage of the total driver actuator travel. The lateral and directional inputs are summed, and the combined output is also limited.

Signal Mixing

The summed and limited input signals are combined in four mixer circuits in the proportions and polarities required to provide command signals for the four swashplate actuator driver servo loops.

Servo Loop

A functional block diagram of the servo loop for one actuator is shown in Figure 17. The active and model servo-amplifiers receive inputs from both the active and the model signal mixers. Thus, the net input to both servo-amplifiers is the average of the active and model mixer outputs. The use of the average results in less variation in actuator commands between DEL control units. It also results in less difference between active and model servo-amplifier currents within a given DEL control unit, thus facilitating servo-amplifier failure detection.

The active servo-amplifier drives an electro hydraulic valve (EHV) in the swashplate driver actuator. See Figure 18. This valve controls the rate of hydraulic flow to one of the three hydraulic cylinders that drive the output ram. Ram position is measured by an LVDT whose output is fed back to the servo-amplifier input, thus closing the servo loop and making ram position proportional to servo-amplifier input voltage.

In a second feed-back loop, the pressure difference (Δp) between the two ends of the ram cylinder is monitored by a LVDT and sent back through a switch-over network as another input to the servo-amplifier. The switch-over network controls the active/on-line status of the channel. In the active channel, the Δp signal is grounded. In the on-line channels, Δp is used as a ram position control signal to keep differential

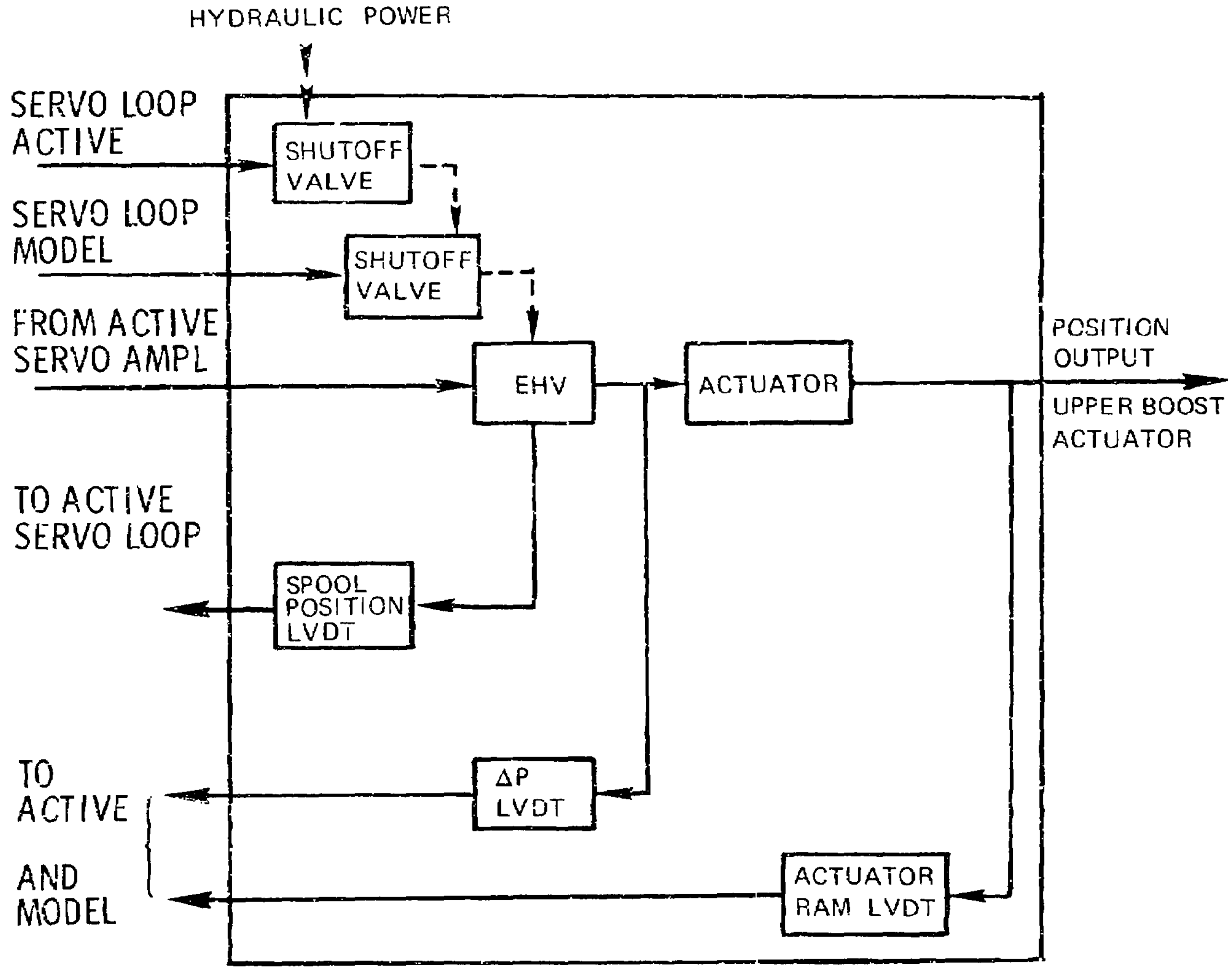


Figure 18. Swashplate Driver Actuator Block Diagram.

pressure at or near zero, thereby preventing ram loading by the on-line channels.

The model servo-amplifier is similar to the active servo-amplifier. It drives a dummy load instead of a valve. The other blocks shown in Figure 17 are parts of a failure detection system discussed later.

Swashplate Driver Actuator (SDA)

The swashplate driver actuator is a dual fail-operate triple-channel electro-hydraulic unit. Each channel consists of a separate c ventional electro-hydraulic actuator. See Figure 19. The status of each actuator, either "active" or "on-line", is determined electrically by the direct electrical linkage (DEL) control system. A two-stage electro-hydraulic servo valve drives the actuator piston. The position of the actuator (main ram) piston and the position of the electro-hydraulic servovalve second stage spool are each monitored by individual linear-variable differential transformers (LVDT).

Only one channel actually controls the output of the triple channel unit. This channel operates fully active and with a relatively high force gain. The two on-line (redundant) channels, although engaged, are incapable of carrying any load. This is accomplished by electrically closing a high-gain lagged-load pressure feedback loop around the actuator using the electro-hydraulic servovalve. If the active channel selected should fail, the solenoid-operated shutoff valve is disengaged, and the channel is isolated from the system and bypassed. Simultaneously the high-gain feedback signal to one of the on-line channels is switched out and that channel becomes active and the controlling channel. In the event an on-line channel fails, only channel disengagement occurs. The authority of the $\Delta\rho$ feedback is intentionally limited. This is done so that the on-line channels will oppose a failed active channel and/or load share as soon as the limit is exceeded.

Each channel has a dual disengagement capability by use of two shutoff valves in series and a dual bypass capability by use of two independent bypass valves, one in the EH servovalve and one associated with the $\Delta\rho$ sensor and the bypass valve.

Failure detection is provided by in-line monitoring of each channel. Three signals are provided to the DEL control unit for this purpose from each of the LVDTs in the separate channels; namely, the piston LVDT, the electro-hydraulic servovalve LVDT, and the differential pressure and bypass valve LVDT. The DEL control unit contains the actuator channel model and the monitoring circuitry for utilizing these signals to properly model the electro-hydraulic servovalve and the actuator servo-loop. See Failure Detection.

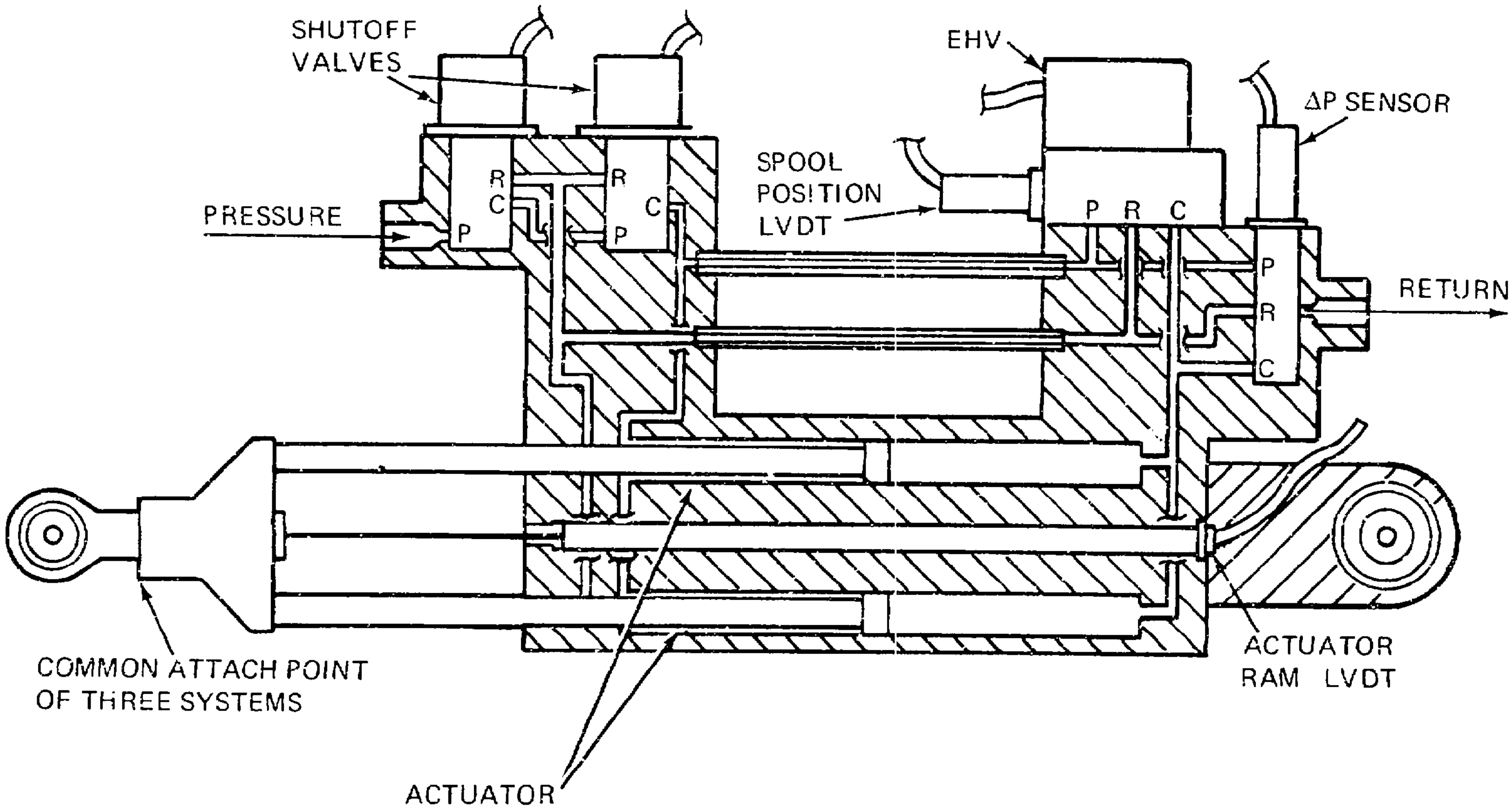


Figure 19. Swashplate Driver Actuator Hydraulic Diagram, Single Channel. (Typical of Three)

There are six solenoid shut-off valves, two per channel, mounted on the actuator assembly. Each valve is a two-stage jet-pipe servovalve fitted with a second stage position monitor and designed with a pressure-off bypass. These valves, when activated, translate electrical signals from the DEL control unit to the hydromechanical movement of the actuator piston. With inlet pressure 250 psi above return pressure, there is no bypass function. With inlet pressure 20 psi above return pressure, the valve is fully bypassed.

There are three differential pressure and bypass valves, one per channel, installed in the actuator assembly. These valves sense the differential pressure across the actuator piston and provide an electrical signal from the LVDT used to nullify the force output of the on-line channels when not in the bypass condition. With input pressure above 250 psi, there is no bypass function. With pressure twenty psi above return pressure, the valve is fully bypassed.

There are three actuator piston position transducer units, one per channel, mounted on the actuator pistons. The LVDT provides an electrical signal to the DEL control unit to null out the electro-hydraulic servovalves.

There are two sets of manifold assemblies per channel. Each set contains the necessary cavities for installation of solenoid valves and differential pressure and bypass valves, mounting surfaces for electro-hydraulic servovalves and internal pressure and return passages for application of hydraulic fluid pressure to these units and the actuator pistons.

FAILURE DETECTION

Operation of the DELS is constantly monitored by failure detection circuits located on the AFCS, signal mixer and servo loop circuit boards. When any failure (except AFCS) is detected, the faulty channel is automatically shut down. AFCS failures are detected and indicated, but they do not shut down the channel as they do not make the channel inoperative.

Most of the failure detectors in the DELS are signal comparators that detect differences in signal levels between the active and model sides of each channel. Differences in mixer outputs are detected by a comparator on the model servoloop board. Failures in the ram position, the differential pressure buffers, the demodulators, the switch-over networks or the servo-amplifiers are detected by comparing servo-amplifier currents. Failures in the electrohydraulic valve (EHV) are detected by the servomonitor, which compares the second stage spool position signal with servo-amplifier current. A failure in

the valve, in the spool position LVDT or in the LVDT buffer or demodulator will result in a disagreement between the servo-amplifier current and the indicated spool position, and a failure will be indicated.

As was previously described, the output of each active mixer is compared with the output of the corresponding model mixer. Active failures are detected by the comparators, and the affected actuators are shut down. However, passive failures do not result in a difference between active and model signals unless there is a control displacement or an AFCS input. By specification, passive failures must be detected with normal stick inputs. These inputs are not sufficient to trip the comparators on the mixer outputs. Therefore, comparators are provided on the limiter outputs to detect passive failures. See Figure 14.

In the servoloops, the same ram position LVDT and the same differential pressure LVDT feed both active and model servo-amplifiers. A failure of one of these LVDTs is not detected by the comparators on the servo-amplifiers because the failure affects both circuits in the same way. To detect these failures, the self-monitor circuit shown in Figure 20 is used. The feedback signal is the sum of an in-phase voltage and an out-of-phase voltage generated by the two LVDT windings. When the LVDT is nulled, these voltages are equal and opposite. When the LVDT is displaced, one voltage increases and the other decreases, but the vector difference between the two remains constant. This constant difference is monitored for failure detection. If either LVDT output winding is opened or grounded, the difference voltage drops to approximately half value; if the two wires are shorted together, the difference voltage becomes zero. Either condition is detected by the undervoltage monitor. If either winding becomes shorted to the input winding, this condition is detected by the overvoltage monitor.

AFCS Failures

Failures in the AFCS inputs or in the input buffers are detected by circuits associated with each voter since a failure affects only one of the three voter inputs. Failures in the multiplexer, voter, demultiplexer, sample hold or ramp generator are detected by the comparators on the signal mixer boards. Such failures cause differences between the active and model inputs.

Hydraulic Pressure Failure

The second-stage spool position LVDT and the differential-pressure sensor LVDT are spring loaded so that they will go to overtravel positions if hydraulic pressure is lost. The

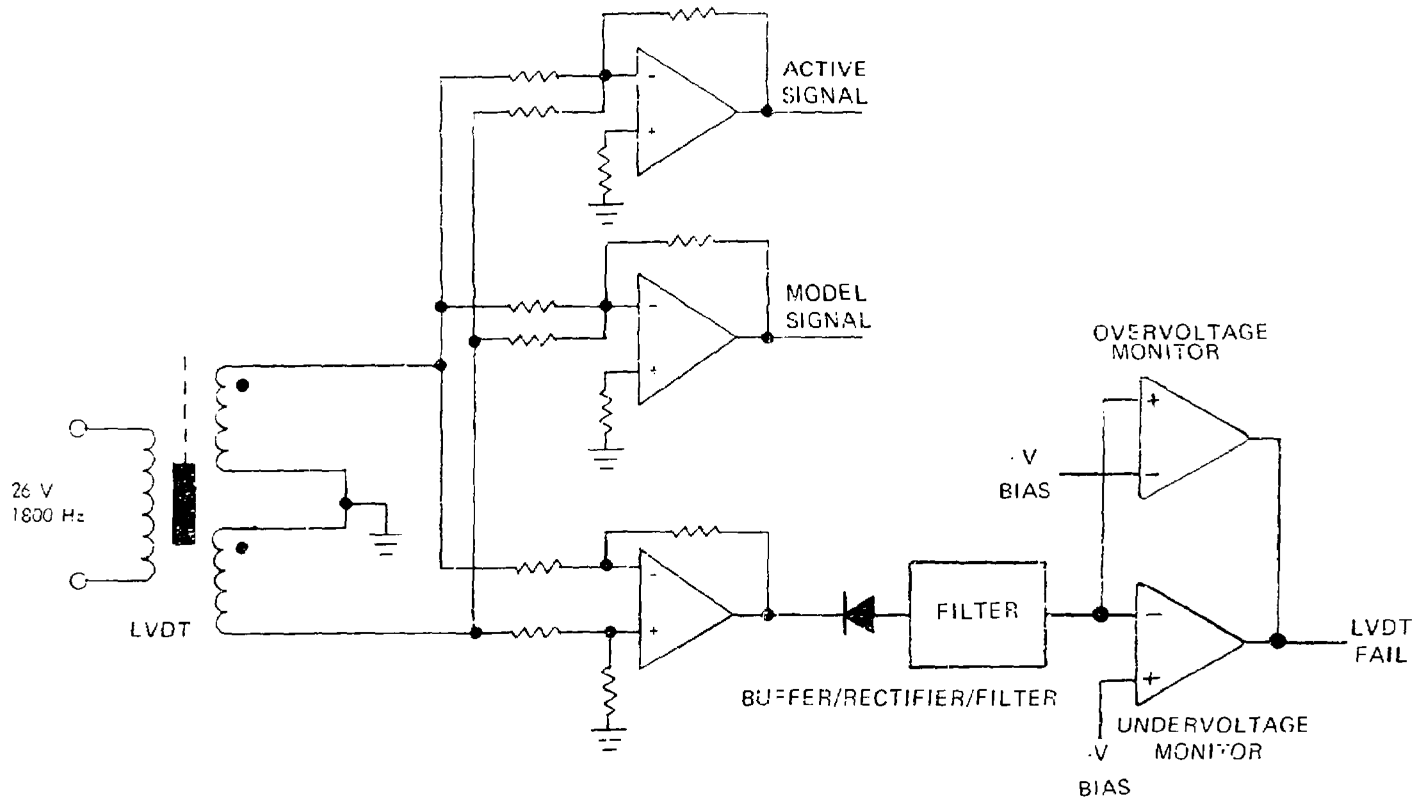


Figure 20. Actuator LVDT Self-Monitor Block Diagram.

resulting abnormally high LVDT outputs are sensed as failure indications by the maximum spool position and maximum differential pressure detectors.

Electrical Power Failure

Loss of any one of the electrical power sources results in the shutdown of one DEL channel. Failure of any of the DC power supplies causes the shutdown logic circuits to indicate a failure. Loss of the 26-V, 1800-Hz LVDT excitation voltage trips the LVDT self-monitors. If 28 VDC is lost, the shutoff valves are directly de-energized.

Actuator Shutdown

When one or more of the failure detectors indicates a failure, the shutdown logic circuits cause the relay drive circuit to de-energize a relay that controls an actuator shutoff valve. There are two such relays and valves, and either valve can shut down one channel of an actuator. The active and model shutdown logic circuits are each connected to both relay drive circuits so if either receives a failure indication, both valves will operate.

Third Failure Shutdown Inhibit

The shutdown logic circuits of all three channels are interconnected to prevent the shutdown of all three channels in any one actuator driver. If any shutdown logic circuit receives a fail signal from both of the other channels, it cannot shut down its own channel. A 2 DEL FAIL signal is generated on each active and model servoloop board.

FAILURE INDICATION TO PILOT

The failure status of the DELS is indicated to the pilot by indicators on the pilot's DELS status panel and the caution panels. The DELS status panel has a failure indicator for each DEL channel along with failure reset switches.

Channel Failure Indication

A block diagram of the failure indication circuit is shown in Figure 21. One side of each of the channel failure indicator lamps is connected to a 28 VDC source; the other side is connected to four relay contacts in parallel: one for each actuator. Since these relays are energized when the actuators are operating, the circuit is open and the lamps are not on. If there is a failure in any output axis of a channel, a relay contact will close and the lamp will turn on. These relays are controlled by the relay drivers on the model servoloop boards.

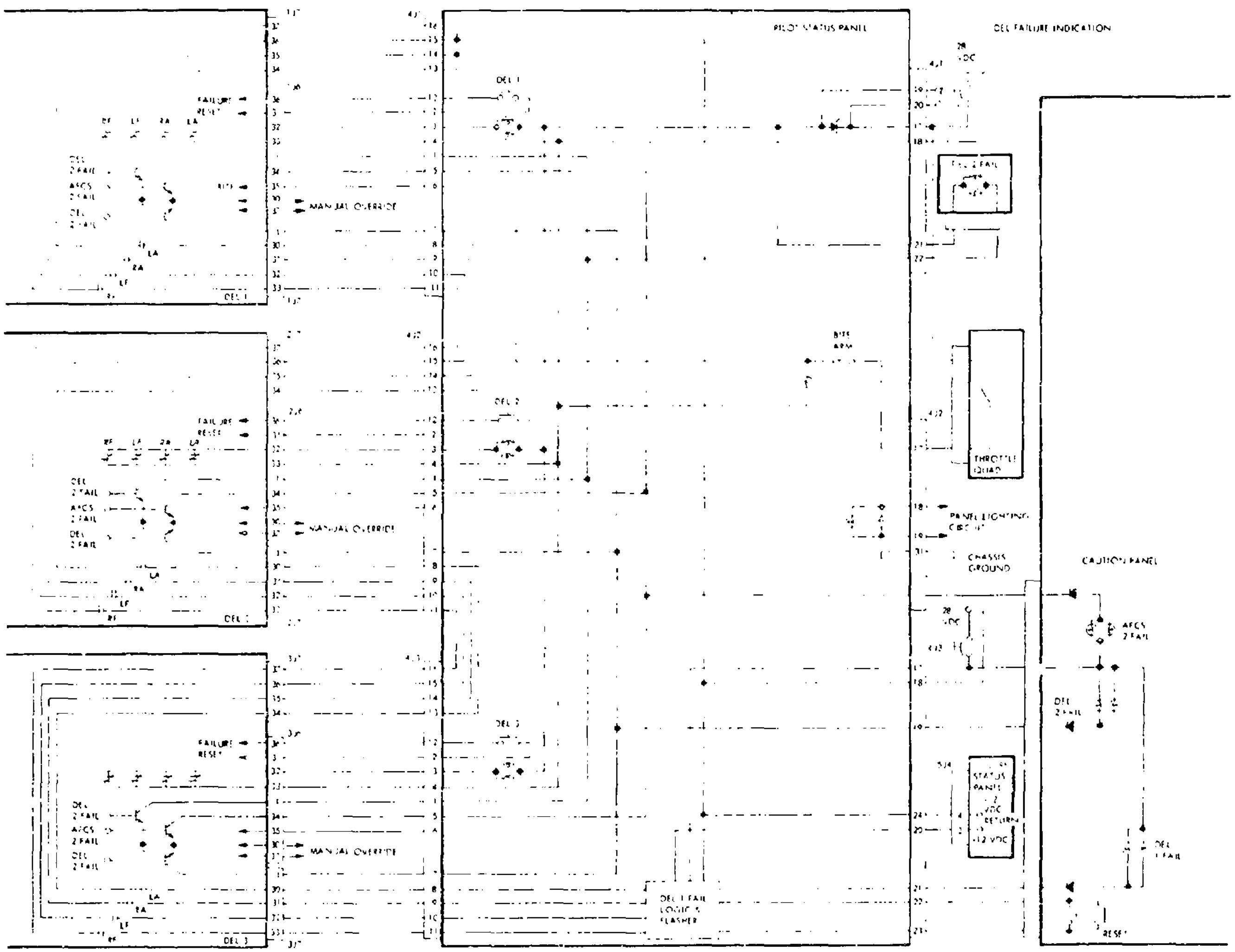


Figure 21. DELS Failure Indication.

Output Axis Failure Indication

Another indication of failures is provided by the first failure indicator on the main caution panel. This indicator is controlled by the single-failure logic and flasher circuit. There are four inputs to this circuit, one from each output axis. The three relays, one in each control unit, in each axis, are connected in series. Since these relays are energized when the actuators are operated, the four inputs to the logic are grounded through the relays. A failure in any channel will remove one of the grounds and the flasher will cause the indicator to flash. Operation of the reset switch on the pilot status panel will turn off both the channel failure indicator and the caution panel first-failure indicator. A subsequent failure in a different output axis will remove a second ground from the flasher, and the indicator will resume flashing. As many as four first failures could occur, one in each output axis, before a second failure occurs in any output axis.

Second Failure Indication

As previously noted, second failure information is generated on each active and model servoloop board for third-failure shutdown inhibit. The second failure information from the four active servoloop boards and from the four model servoloop boards are separately combined, resulting in two second-failure signals per control unit. As shown in Figure 21, these signals control transistor switches which are wired in parallel to the low side of two second-failure indicators, one on the main caution panel and one on the auxiliary caution panel. The indicators are connected to separate 28-VDC supplies so an indication will be provided unless both supplies are lost. Loss of power in one control unit will prevent closing the switches in that unit, but power would have to be lost in all three units to prevent the illumination of the indicator.

AFCS OFF Indication

AFCS OFF indication is similar to second-failure indication. Transistor switches in each control unit are wired in parallel to the low side of an indicator on the caution panel. If any one of the three control units has an AFCS OFF signal, the indicator will be illuminated.

DELS Status Indication

DELS status indication is provided by indicators on the DELS status panel shown in Figure 21. There are eighteen indicators and a reset switch for each channel. Operation is the same in each channel. One channel will be described.

SDA Indicators

Four SDA indicators are provided, one for each actuator. An indicator will turn on if the shutdown logic on either an active or a model servoloop board indicates a failure. The indicators show which swashplate driver actuator locations are shut down.

Input Indicators

Four indicators are provided, one for each input axis. An indicator will be turned on if the comparator on either the active or a model stick/mixer board detects a failure. An input failure will result in illumination of one input indicator and four SDA location indicators.

Servo Indicators

Indicators are provided for the mixer output comparator, the servo-amplifier current comparator, the differential pressure LVDT self-monitor and the ram LVDT self-monitor. If any one of these detects a failure, the corresponding indicator and one of the SDA location indicators will be illuminated. Indicators are not provided on the maximum differential pressure or maximum spool position detectors because these detectors will indicate a failure whenever an actuator channel is shut down and are therefore redundant to the SDA location indicators.

Active Channel Indicators

Active channel indicators are provided for each of the four output axes. A lighted indicator identifies the DELS channel that is in control of that particular actuator. These indicators are not latched; therefore, they are not affected by operation of the indicator reset switches.

Indicator Reset Switches

The failure indicators will turn on and are electronically latched whenever there is a failure. They will stay on, even if the pilot successfully resets the failure, until the indicator reset switch is operated.

Power Supply

The indicators are energized by a power supply in the failure status panel excited by 115V, 400 Hz. This is done to avoid the possibility of one failure affecting all three channels, which might exist if all three DEL 28-VDC supplies were brought into the failure status panel.

BUILT-IN-TEST EQUIPMENT (BITE)

Test equipment built into each DEL control unit provides a fast, easy-to-use means, on a go/no-go basis, for checking the condition of the control unit. It can also be used, under manual control, in conjunction with the Preflight Test Set or the DEL Subsystem Test Bench as an aid in fault isolation.

DEL Self-Test Operation

The built-in test equipment (BITE) is shown in block diagram form in Figure 22. One BITE panel is used to control the BITE in the DEL control units. The DEL to be tested is selected by a switch on the BITE panel. Through the DEL Select line, a relay in the selected DEL is energized and supplies power to the BITE circuits. Testing is started by the test initiate signal from a pushbutton switch on the BITE panel. From this point on, the testing proceeds automatically.

The BITE control logic starts and stops the testing and controls the sequencing of the test operation. A binary counter counts clock pulses. Counter output is decoded to produce test number signals. The test signal generator applies the test signals (DC voltages, grounds, logic levels) required for each test to BIT insertion points in the DEL circuitry. The fault detector monitors specified points for each test fail signal if the correct logic levels are not present.

When a test sequence is successfully completed, as indicated by the last test +1 signal from the test number decoder, the BITE control logic stops the counter and generates a GO signal. This signal illuminates a lamp on the BITE panel to signal successful test completion. If a test is failed, the test fail signal causes the test sequence to stop. The GO lamp on the BITE panel is not lit (NO-GO condition), and test number indicators display the number of the test that failed.

Boxed Controls Test

The BITE also provides for a boxed controls test used for checking the tracking or balance between the three DEL channels. There are eight tests in the sequence, two for each actuator channel. Progression from one test to the next is manually controlled from the BITE panel. In each test, the inputs to the system are derived from specified hard-over movements of cockpit controls. In the two on-line control units, the BITE monitors the Δp LVDT signal. A test failure is indicated if this signal rises above a specified level.

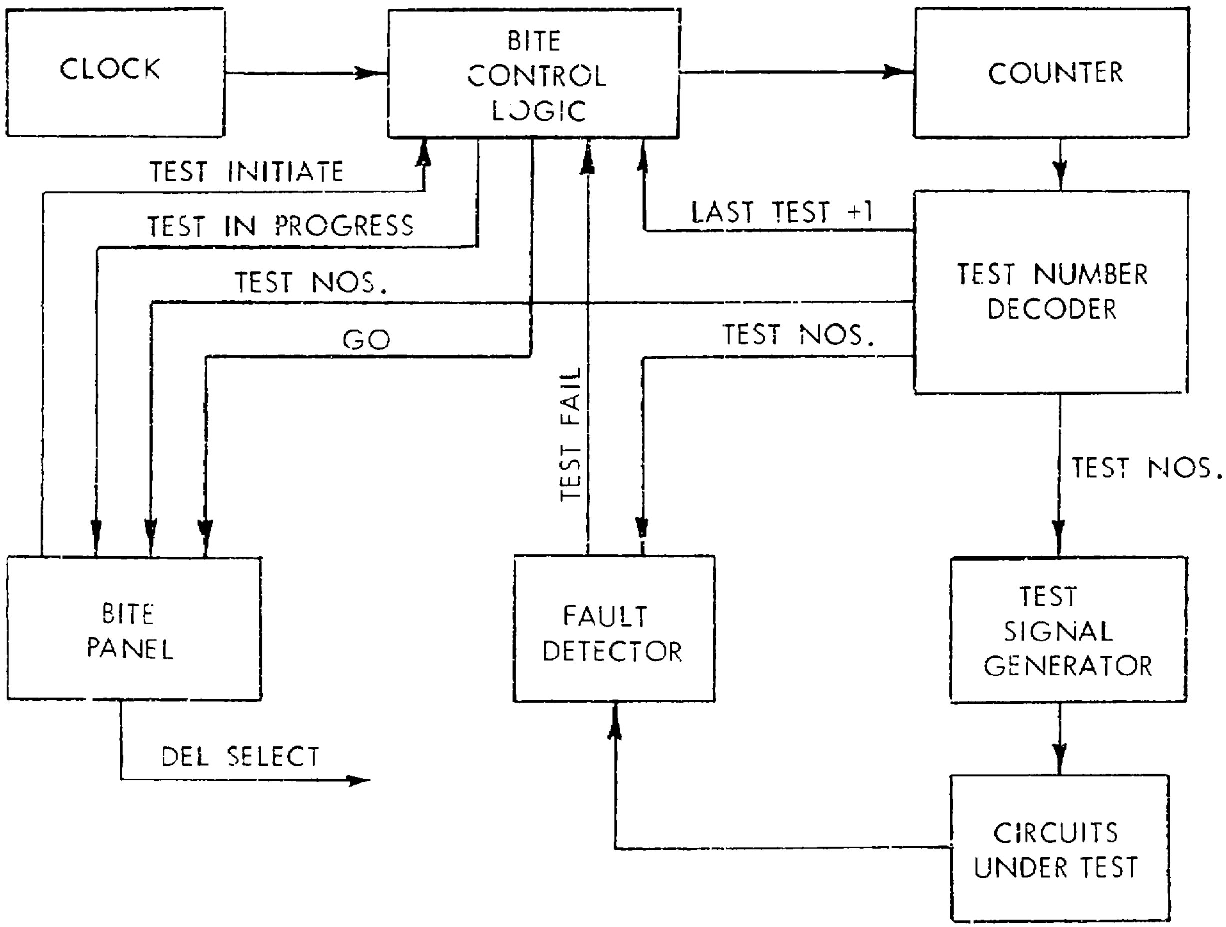


Figure 22. DELS BITE Functional Block Diagram.

DELS TEST PROGRAM

Testing of the DELS followed an orderly progression. Principle test programs in chronological order were:

1. Acceptance Tests - performed by subcontractor to verify that the fabricated equipment met the procurement specifications.

2. Limited Qualification Tests - environmental testing to substantiate "flight clearance" qualification.

3. Integration Tests - a complete systems test. DELS was installed on "iron bird" integration test stand to verify overall system performance.

4. Ground and Flight Tests - feasibility demonstration on the model 347 test vehicle.

The detail report material in the following paragraphs will be devoted to the results of integration and ground/flight testing. Only the major findings of acceptance and flight clearance qualification will be presented.

Acceptance Tests

Functional tests of the DELS equipment were performed at ambient conditions to demonstrate acceptability. Equipment operation within specified tolerances was used as a condition of acceptance.

Stick Position Transducer Acceptance Tests

Each transducer was tested to demonstrate specification compliance over the total stroke. Some of the transducers had marginal performance in terms of scale factor, tracking and cross coupling. Acceptance was granted after system tests showed that the marginal conditions had no adverse effects on end-to-end system operation.

Control Unit Acceptance Tests

Closed-loop operation of each control unit was verified using special support equipment to simulate transducer inputs and actuator feedback. The tests demonstrated satisfactory control unit performance. Gains, limits and failure detection characteristics met the specification requirements for each control unit delivered.

DELS Panels Acceptance Tests

Passive and active tests were conducted on each of the three types of system panels (BITE, Failure Status and Pilot Status) to verify acceptable performance.

Swashplate Driver Actuator Acceptance Tests

Tests were conducted to verify acceptable operation of the Δp sensor and the driver actuator assembly. Electrical and hydraulic parameters were shown to be within the specified tolerances for each Swashplate Driver Actuator.

DELS System Acceptance Test

DELS components were integrated into the total system configuration, and tests were conducted to verify end-to-end operation. Failure detection capability was determined by performing BITE functions and by observing panel indications resulting from induced system failures. System response was evaluated by applying inputs to the stick position transducers and measuring the resulting actuator movement. The two DELS systems used in the ATC program met all the specification criteria.

Flight Clearance Qualification Tests

Environmental tests were performed to demonstrate that the DELS components function safely in the design's operational environments or after exposure to the environments. Satisfactory operation at the environmental conditions specified was used as a basis for granting flight clearance. Stick position transducers were previously qualified under another program; no additional tests were made. A summary of the tests performed on control units, panels and swashplate driver actuators is given below.

Control Unit Flight Clearance Tests

Temperature/Altitude tests were conducted in accordance with MIL-STD-810B, Method 504. Control unit performance was satisfactory during and after exposure to the environment and no deterioration was noted.

Humidity tests were conducted in accordance with MIL-STD-810B, Method 507. Discrepancies were noted during the post-humidity operational test. The discrepancies were isolated to two causes: 1) insufficient bonding of circuit board conformal coating allowed moisture to enter and form low impedance paths, 2) some of the metal film resistors failed due to electrolysis during the high humidity exposure.

The humidity tests were repeated after circuit boards had been coated using an improved coating process. Discrepancies occurred after the retest, but it was determined that they were caused by failures of eight of the 1,114 metal film resistors; no discrepancies were attributed to failure of the conformal coating. Vibration tests were performed in accordance with MIL-STD-810B, Method 514.1. Vibration endurance frequencies were selected based on results of resonance search tests. No discrepancies occurred during vibration endurance at the selected frequencies.

DELS Panels Flight Clearance Tests

The pilot status panel, BITE panel and failure status panel were tested to demonstrate operation in the temperature/altitude and vibration environments specified in MIL-STD-810B. No panel discrepancies occurred in either of the environments.

Humidity tests were conducted in accordance with Method 507 of MIL-STD-810B. Several discrepancies occurred during the post-humidity performance tests conducted on the panels. Approximately fifty percent of the light-emitting diodes (LEDs) used on the BITE panel and the failure status panel failed to illuminate. All of the LEDs suffered various degrees of discoloration that was attributed to moisture penetration. Special tests were conducted on another type of LED which was recommended by the manufacturer for replacement of the failed LEDs. The replacement LEDs showed no signs of deterioration after high humidity exposure.

The humidity tests of MIL-STD-810B were rerun to provide a complete evaluation of the new LEDs. During the post-humidity performance test, fifteen of the forty-eight LEDs failed to illuminate. Although there was no discoloration of the lenses, it was apparent that moisture penetration caused the thirty-three percent failure rate. Because no MIL Standard LEDs were available at the time, an engineering investigation was initiated to find a suitable moisture-resistant LED to meet the humidity test requirements. Until the LED problem is resolved, the failure status panel and the BITE panel fail to meet the humidity performance requirements.

Swashplate Driver Actuator Flight Clearance Tests

Flight clearance tests were conducted to demonstrate that the swashplate driver actuator functions safely at design limit conditions. A summary of the tests is given below.

The humidity test of MIL-STD-810B, Method 507, was performed. The actuator showed no deleterious effects from the humidity performance test. Insulation resistance of the electrical connectors was lower than required, but this was due to

moisture penetration through open spare pins. The insulation resistance test was repeated with dummy pins inserted in the spare locations. The retest was successful. High and low temperature tests were conducted to demonstrate actuator operation at -25° and +160°F. No degradation in performance occurred at either temperature extreme.

The Impact Shock Test of MIL-STD-810B, Method 516.1, was conducted. Post shock tests were performed, and the actuator functioned satisfactorily.

Vibration and fatigue/wear tests were performed. Life-cycle testing was conducted at a vibration frequency of 30 Hz, chosen by Boeing Vertol. The actuator performed satisfactorily during the vibration and fatigue/wear test.

Disassembly and inspection of the actuator assembly revealed that no damage resulted from the environmental tests described above.

DELS System Flight Clearance Tests

Tests were conducted at ambient conditions to determine the frequency response and susceptibility to noise and voltage transients of the integrated DELS system. The tests were divided into three phases as described below.

Frequency Response

Gain and phase shift of the DELS system were determined by applying inputs of constant amplitude over a range of frequencies and measuring output amplitude and phase at the actuator.

Frequency response and phase characteristics met the specification criteria with three DELS channels powered and with one DELS channel powered.

Noise Susceptibility

Interconnecting cables and signal leads of the DELS system were subjected to electromagnetically coupled relay transients to determine system noise susceptibility. Actuator displacements were within required limits, but false status indications occurred when noise was coupled into two of the seven system cables. Buffering of LED indicator circuitry was the recommended solution, but action was deferred until EMI tests of the aircraft installation.

Voltage Transient Tests

Voltage transients of 500-700 volts peak-to-peak were superimposed on system 28-VDC input power lines to determine their

effect on actuator displacement and failure status indications. Actuators remained within required limits and no failure indications occurred.

DELS Systems Test Configurations

A three step plan to progress from initial flights to a pure fly-by-wire flight test demonstration resulted in three distinct control system configurations. In both the integration test and the flight test, the configurations were identified as Phases I, II, and III.

Phase I - DELS Open Loop

This was a temporary, first-flight configuration. Flight control was by means of the existing mechanical system. The DELS was functional except that the output of the swashplate actuator was disconnected. Instrumentation recorded the SDA output for comparison with the mechanical system.

Phase II - Fly-By-Wire with Mechanical Backup

This configuration was also a temporary phase in which the copilot served as a safety pilot. This was accomplished by making both the mechanical and the fly-by-wire operative.

The pilot-copilot control synchronization linkages were disconnected. Pilot control commands operated the fly-by-wire system. The output ram of the fly-by-wire driver actuator drove the upper boost swashplate actuators and back drove the mechanical system. To operate in the fly-by-wire mode, it was necessary to bypass the lower boost actuators of the mechanical system. To operate the mechanical controls, it was necessary to bypass the SDAs.

The normal mode in Phase II was fly-by-wire. An automatic and a manual means of transfer to mechanical backup was provided. The automatic failure detection and transfer mechanism utilized signals from four position sensors located to measure swashplate actuator motion. These signals were compared with signals representing cockpit control commands. Errors beyond a fixed threshold amount would cause automatic transfer to mechanical backup.

Phase III - Fly-By-Wire Without Mechanical Backup

No mechanical backup was available in the Phase III configuration. The only means of flight control was the fly-by-wire. Both the pilot and copilot control drove the stick position transducers. The output ram of the SDAs drove only the control valves of the upper boost actuators.

DELS Integration Tests

Functional testing of the DELS used an "iron bird" integration test stand. A photograph of the stand is shown in Figure 23. The installation of DELS simulated the aircraft as far as practical. "Cabin racks" were used to mount the control units, and duplicate aircraft wire bundles were employed. The aircraft mechanical system from cockpit to upper controls was included. Laboratory electrical and hydraulic power supplies were employed, but switching and transfer mechanisms were flight hardware.

INTEGRATION TEST RESULTS

Test results are reported in Reference 1. The results are summarized as follows:

1. Complete DELS functions and all failure modes were simulated. System function was proper. BITE was verified.

2. Static and dynamic performances were measured in the three configurations, Phases I, II and III, planned for flight testing. Selected frequency response data is shown in Figures 24 through 28. Performance was found to be satisfactory. The principle success criteria for the frequency response was that the amplitude ratio of the SDA should be flat within ± 3 db at 9 Hz and less than seventy-five degrees phase shift. It was determined that the frequency response was adequate for the high-frequency automatic flight control system functions. All configurations met the criteria.

3. The static performance data of gain between the pilot controller input and the output of the upper boost actuator is shown in Table 1. This data shows that the electrical system is equivalent to the mechanical system it replaces; a high priority success criteria for the fly-by-wire controls. Measurements made on the aircraft in ground tests further substantiated this conclusion.

4. The failure detection and transfer mechanism for the Phase II configuration was made to function satisfactorily.

5. AFCS interface circuits were satisfactorily tested by inserting simulated signals.

6. DELS response was checked and found satisfactory for voltage and frequency variations expected in the test

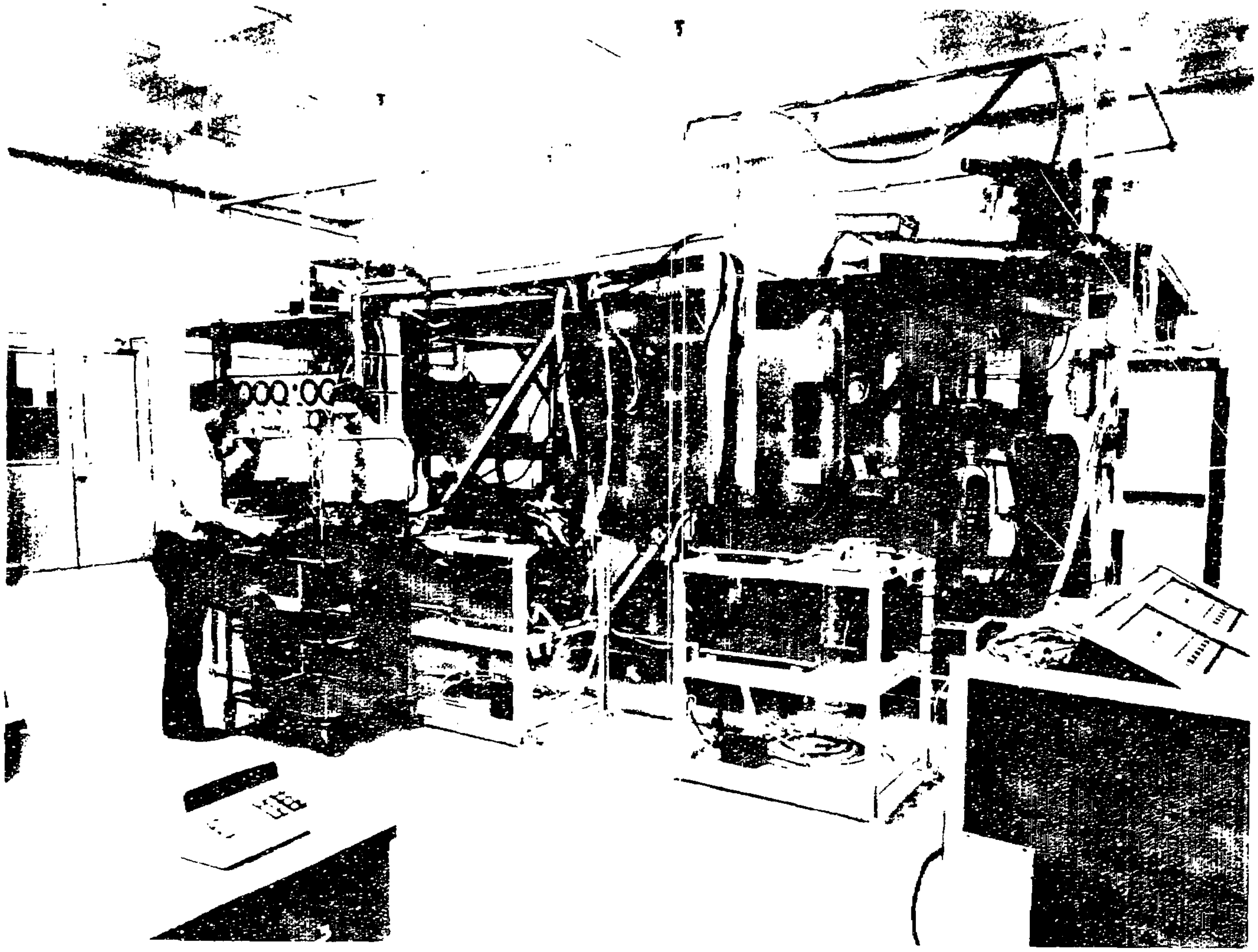


Figure 23. Integration Test Stand.

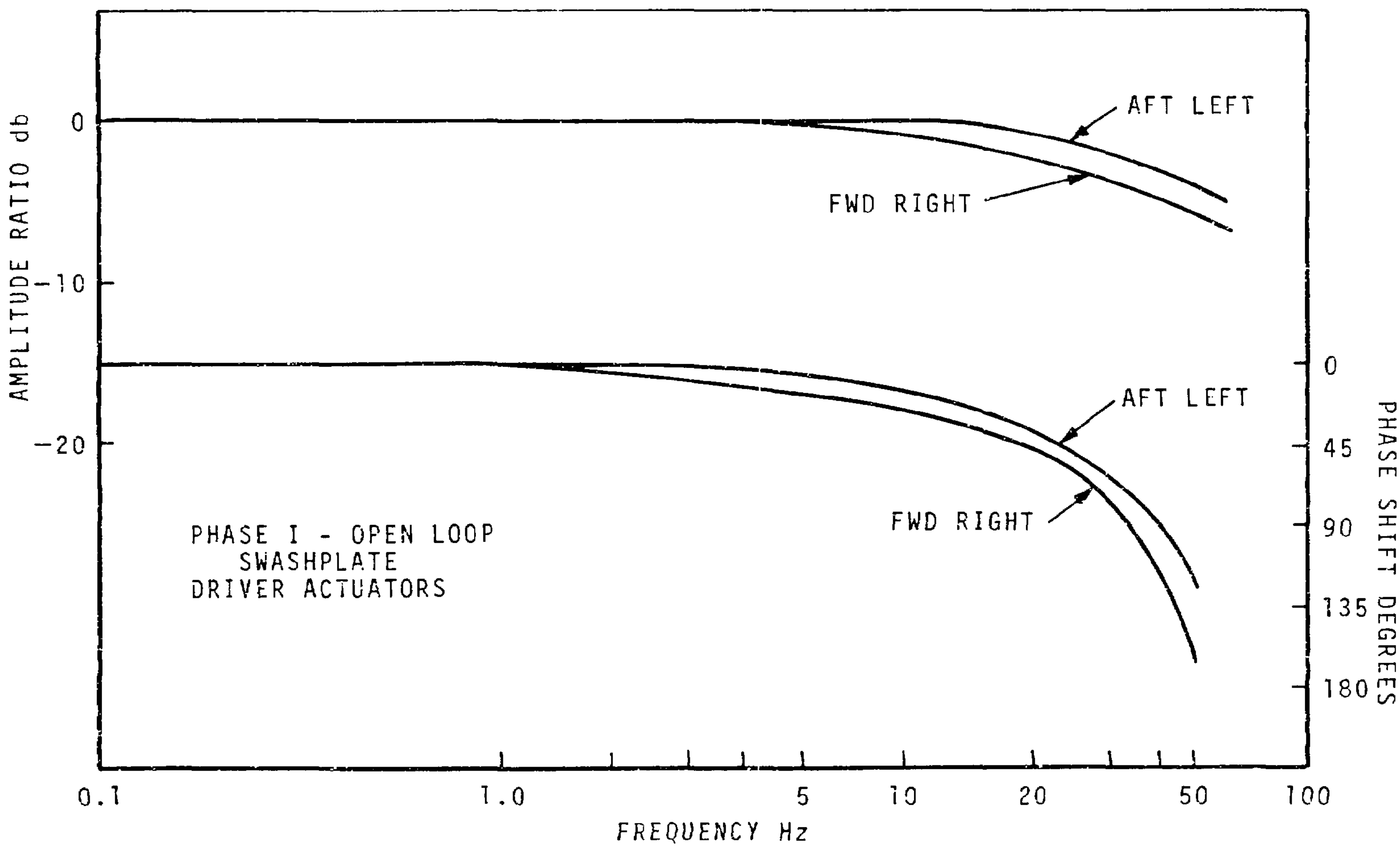


Figure 24. Typical Frequency Response - Integration Test Data - DELS Open Loop.

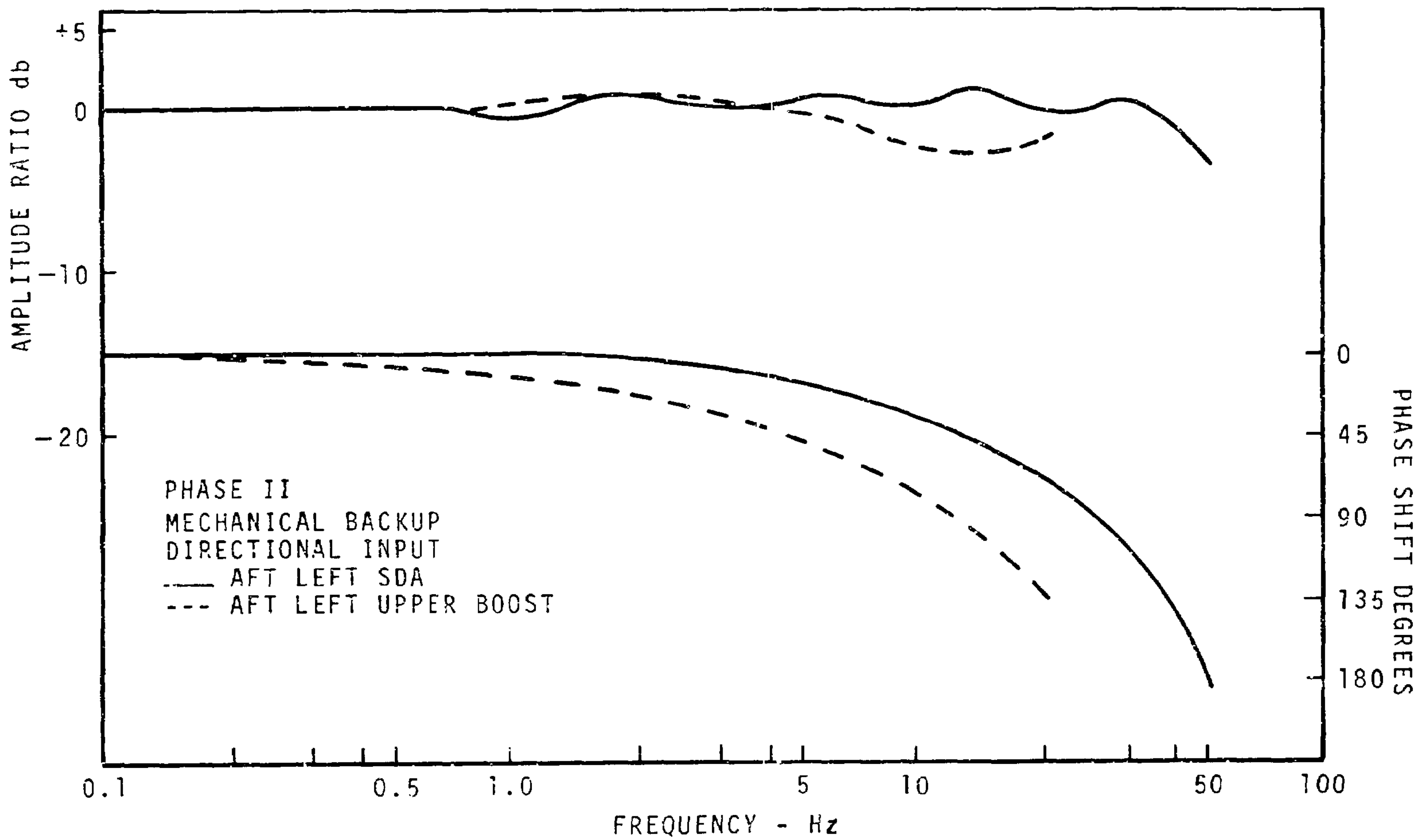


Figure 25. Frequency Response - Integration Test Data - Mechanical Backup.

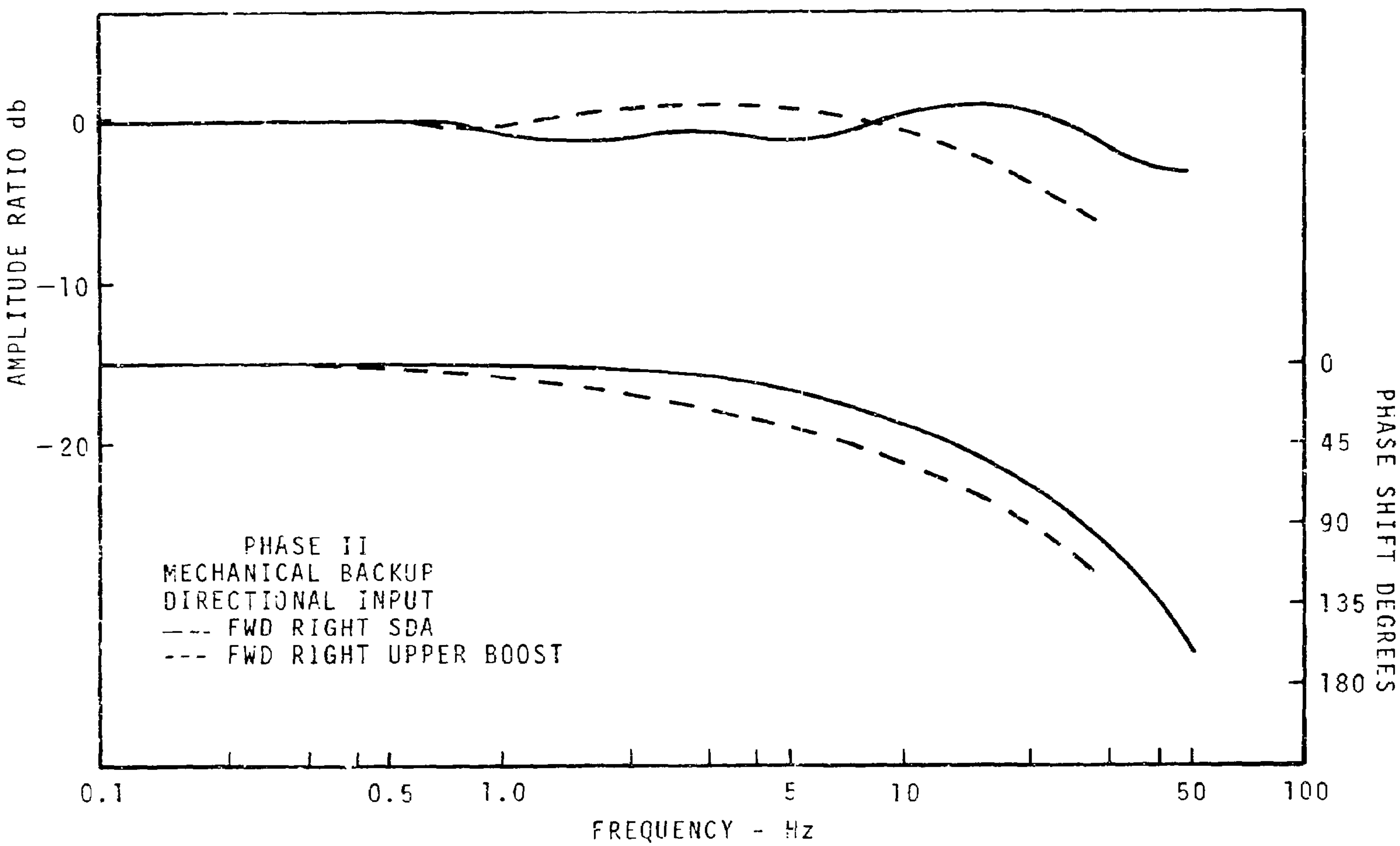


Figure 26. Frequency Response - Integration Test Data - Mechanical Backup.

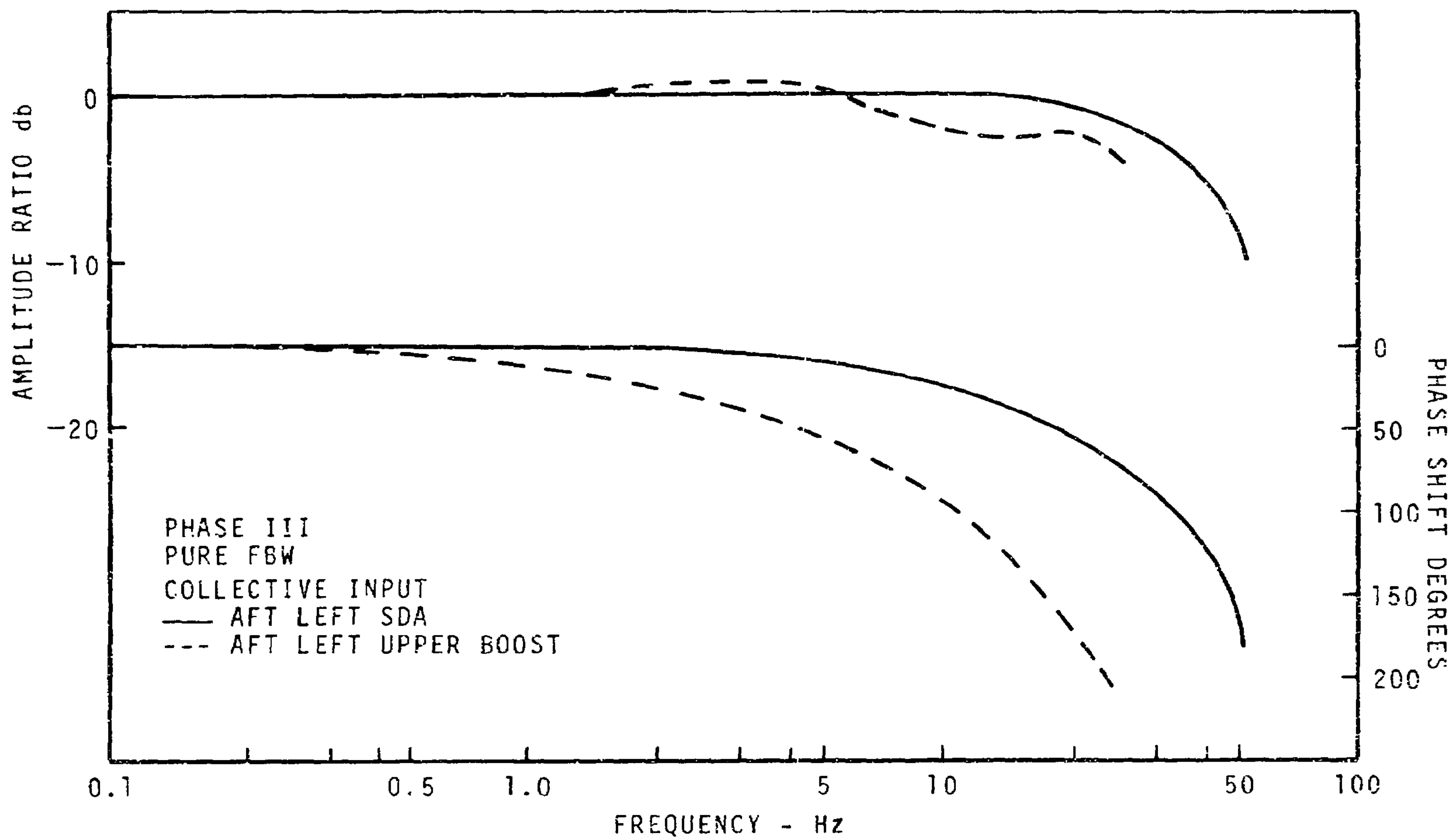


Figure 27. Typical Frequency Response - Integration Test Data - Pure Fly-By-Wire.

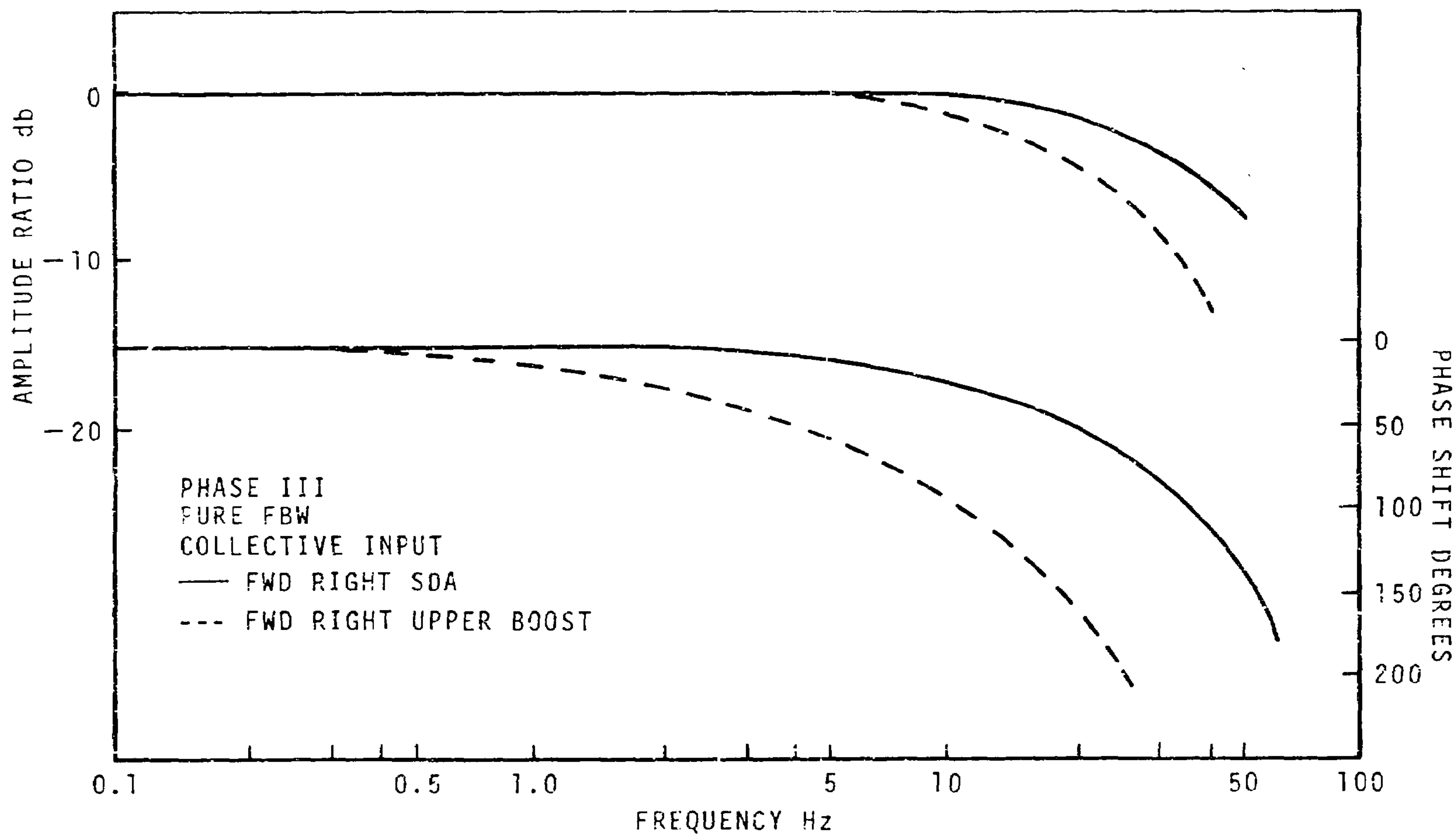


Figure 28. Typical Frequency Response - Integration Test Data - Pure Fly-By-Wire.

TABLE 1. STATIC GAIN COMPARISON -- FLY-BY-WIRE AND MECHANICAL FLIGHT CONTROLS

CONTROL	UPPER BOOST ACTUATOR	GAIN RATIO IN/IN INTEGRATION TEST STAND		AIRCRAFT SYSTEM LIMITS	
		MECHANICAL	DELS*	LOW	HIGH
COLLECTIVE	Fwd. Right	0.365	0.356	0.355	0.369
	Fwd. Left	0.355	0.360	0.355	0.369
	Aft Right	0.339	0.337	0.328	0.342
	Aft Left	0.340	0.341	0.328	0.342
LONGITUDINAL	Fwd. Right	0.100	0.097	0.097	0.101
	Fwd. Left	0.099	0.095	0.097	0.101
	Aft Right	0.093	0.090	0.090	0.904
	Aft Left	0.097	0.094	0.090	0.094
LATERAL	Fwd. Right	0.443	0.439	0.438	0.456
	Fwd. Left	0.444	0.430	0.438	0.456
	Aft Right	0.431	0.424	0.422	0.440
	Aft Left	0.424	0.434	0.422	0.440
DIRECTIONAL	Fwd. Right	1.062	1.064	1.060	1.109
	Fwd. Left	1.060	1.044	1.060	1.109
	Aft Right	0.979	0.987	0.970	1.010
	Aft Left	0.972	1.006	0.970	1.010

*Corrected for mechanical advantage (SDA to Upper Boost) difference between aircraft and test stand.

vehicle. A performance check using the backup batteries was performed.

7. Test procedures and baseline data for ground and flight tests were established.

8. Pilots and aircraft test personnel were checked out on DELS operating procedures.

9. System hours accumulated during integration testing were:

Phase I - Open Loop:	52 hours
Phase II - Mechanical Backup:	245 hours
Phase III - Pure Fly-By-Wire:	43 hours
Total	340 hours

10. Test hours accumulated on individual DELS Control Units were:

S/N 73001	198 hours
73002	236
73003	228
73004	69
73005	68
73006	67
73008	69
73009	28
Total	963 hours

11. There were only a few failures during integration, ground, and flight tests. A complete list is provided in a later paragraph. See DELS Failure and Malfunction Record.

Ground Tests

The flight program was preceded by ground tests, including system functional checks and Electromagnetic Compatibility/ Radio Frequency Interference testing. The results are reported in the Reference 2 document. Abbreviated descriptions of pertinent tests and data are presented herein.

After making the fly-by-wire functional test in the aircraft, static checks were made to show that the fly-by-wire was equivalent to the mechanical system and that there were no basic changes to control kinematics from one test configuration to the other.

Hysteresis Checks

Typical "X-Y" plots, as shown in Figure 29, show no detrimental changes in kinematics for the various configurations. Note that least hysteresis was obtained with the fly-by-wire system, pure DELS, in the figure. This is thought to be one reason why the pilot commented that "it flew like a new, well-adjusted mechanical system".

Static Performance, Control Position Data

Static performance tests were made to obtain control position versus upper boost actuator position in order to compare the DELS and the mechanical system. Typical test data is shown in Figures 30 and 31. Plots of movements in each control axis with the other three axes at neutral (rig pins in) are presented.

The measured gains are within limits. Comparison of the mechanical system with the fly-by-wire indicates that DELS is an accurate electrical equivalent of the mechanical controls.

EMC/RFI

EMC/RFI testing was conducted to ensure that flight safety was not degraded by RFI in or between the basic aircraft and its instrumentation systems and the DELS. The tests were accomplished in the Phase I configuration, initially with electrical and hydraulic power supplied by auxiliary power unit (APU) and repeated during unbladed rev-up with aircraft engines. Selected conditions were again repeated during bladed rev-up following installation, relocation or modification of some DELS equipment.

Minor actuator disturbances were noted during these checks, none of which concerned safety or flight. For the most part, the minor disturbances were transitory and were made a matter of record, but no fix was incorporated. One exception was that filters were installed to prevent "DELS Failure Status" lights from flashing because of some electrical power switching.

Transfer to Mechanical Backup

Preliminary to and during Phase II, the reversion from DELS operation to mechanical controls was checked in both the automatic and manual modes. Manual reversions were performed with DELS engaged by depressing the DELS release switch on either the pilot's or copilot's cyclic grip. Automatic reversion was activated by a test input to the FDTA. The reversions were always completed as intended.

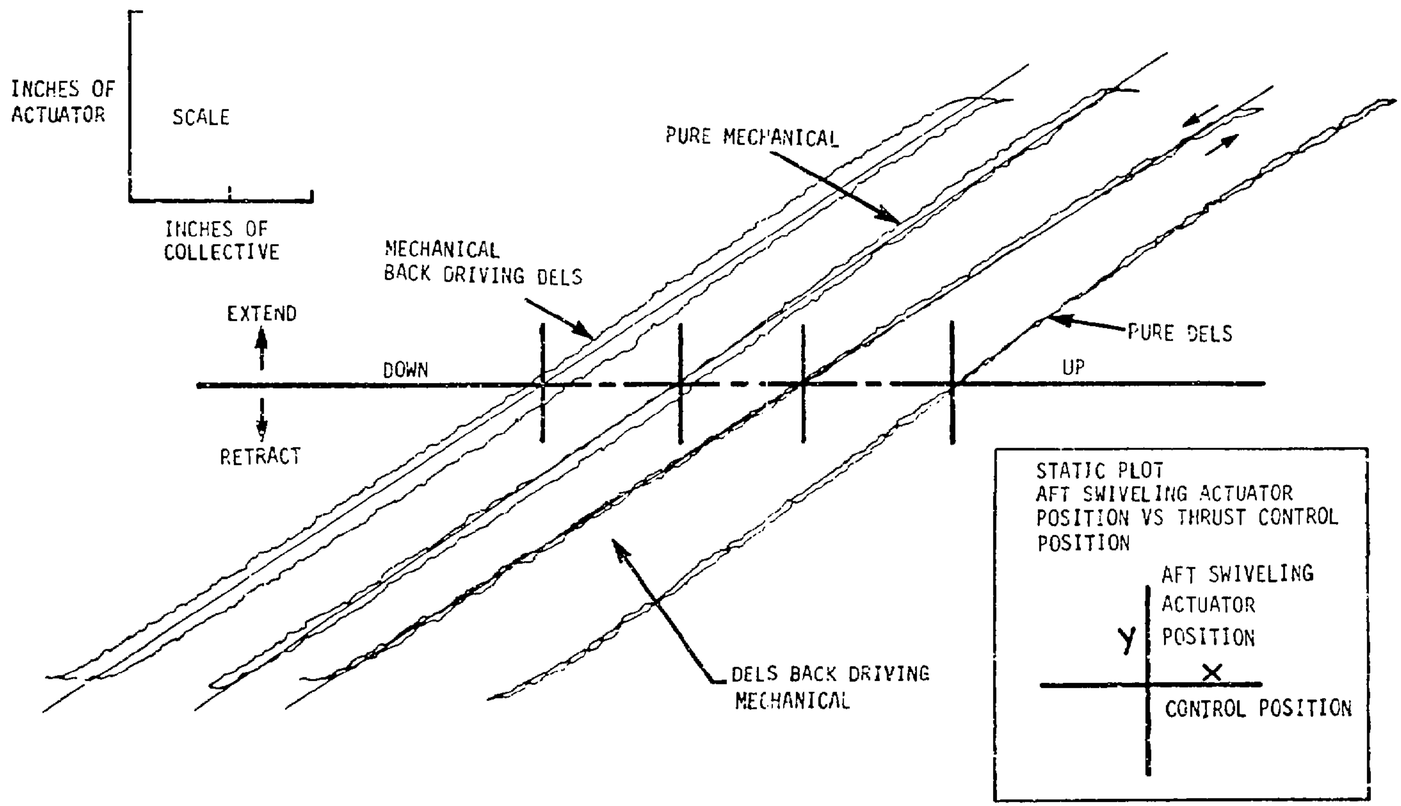


Figure 29. Hysteresis Plots, Fly-By-Wire and Mechanical Systems.

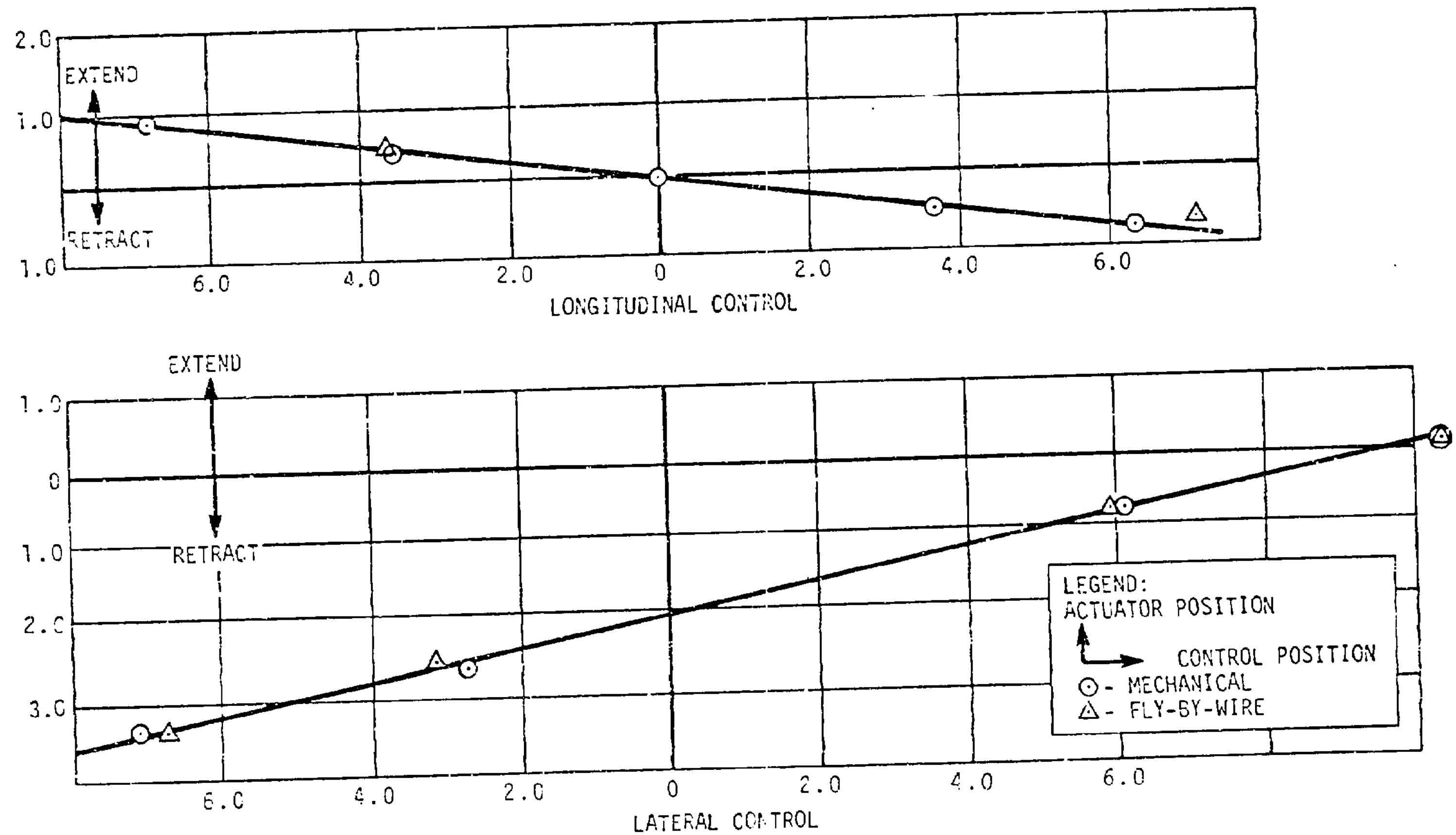


Figure 30. Static Plots, Control Position Versus Forward Swiveling Actuator. Position

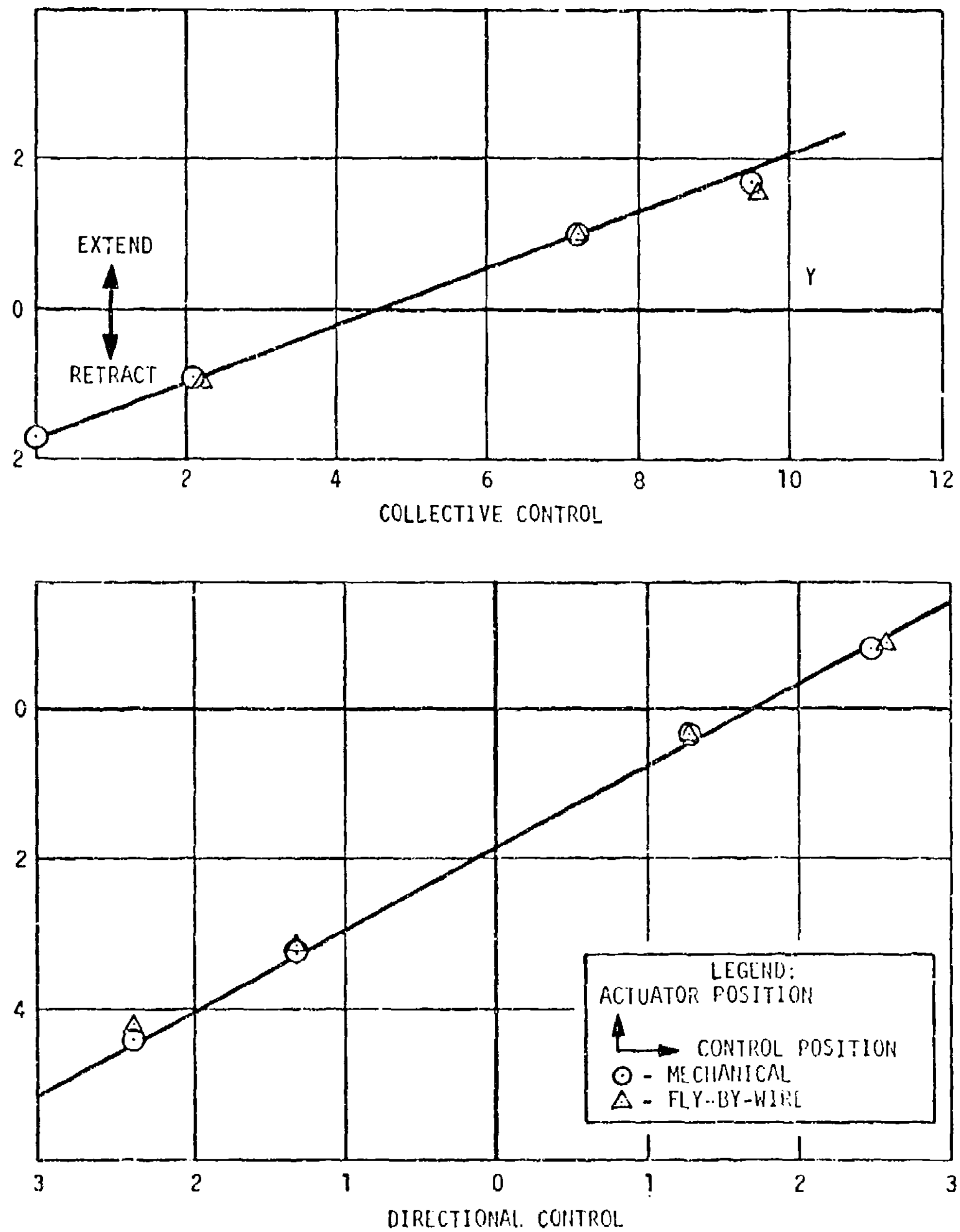


Figure 31. Static Plots, Control Position Versus Forward Swiveling Actuator Position.

Flight Tests

The flight test evaluation was conducted in three phases according to plan. The flight time per phase and the specific tests performed are summarized in Table 2. Pertinent test results are presented in brief in the following paragraphs. The Flight Test Report is contained in Reference 2.

Phase I - DELS Open Loop

The objective of this test was to establish the fidelity of the fly-by-wire related to the mechanical system. The aircraft was flown using the mechanical system with the DELS operational but with the outputs of the SDAs disconnected. Instrumentation recorded the mechanical and electrical system responses. The reduced data showed nearly perfect fidelity between the two systems. Representative data is shown in Figures 32 and 33.

Airborne EMC/RFI

No problem with DELS and related equipment was observed. In the test aircraft, systems were operated, such as windshield wipers, anti-collision lights, windshield, anti-ice, pitot heater, engine inlet anti-ice, IFF equipment, VHF and FM radios. The aircraft was operated near an operating HF radio in another helicopter and was directed by PHL radar over a mark beacon, and near a commercial radio station antenna tower.

DELS First and Second Failures

Since DELS was open loop, the pilot's qualitative assessment of aircraft response following failures could not be determined. However, all failure warning and status indications were correct. It should be noted that the DELS design prevents a third channel shutdown after [illegible] channels have failed.

AFCS First, Second and Third Failures

No pilot assessment of aircraft response since DELS was open loop. All failure warning and status indications were correct.

Phase II - DELS with Mechanical Backup

In this phase, the pilot controlled the helicopter with the fly-by-wire system, and the copilot could control the helicopter using the mechanical controls.

TABLE 2. SUMMARY OF DELS FLIGHT TESTING

TEST PHASES	TIME	TEST PERFORMED
I - DELS OPEN LOOP	3:35	• Standard Flight Profile consisting of: Hover, Transition, Climbing Turns, Control Frequency Sweeps, Control Input Pulses, Control Reversals, Sideslips, Autorotation, Speed Sweeps, Sideward Flight, Rearward Flight • EMC/RFI • DELS 1st & 2nd Failures • AFCS Interface 1st, 2nd, & 3rd Failures
II - FBW WITH MECHANICAL BACKUP	7:12	• Standard Profile (above) • Control Sensitivity • DELS 1st Failure • AFCS Interface Failures • DELS Single Channel Flight • Reversion to Mechanical Backup
III - PURE FLY-BY-WIRE	8:02	• Standard Profile (above) • Demonstration Flights • Army Flight Evaluation
TOTAL TIME	18:49	

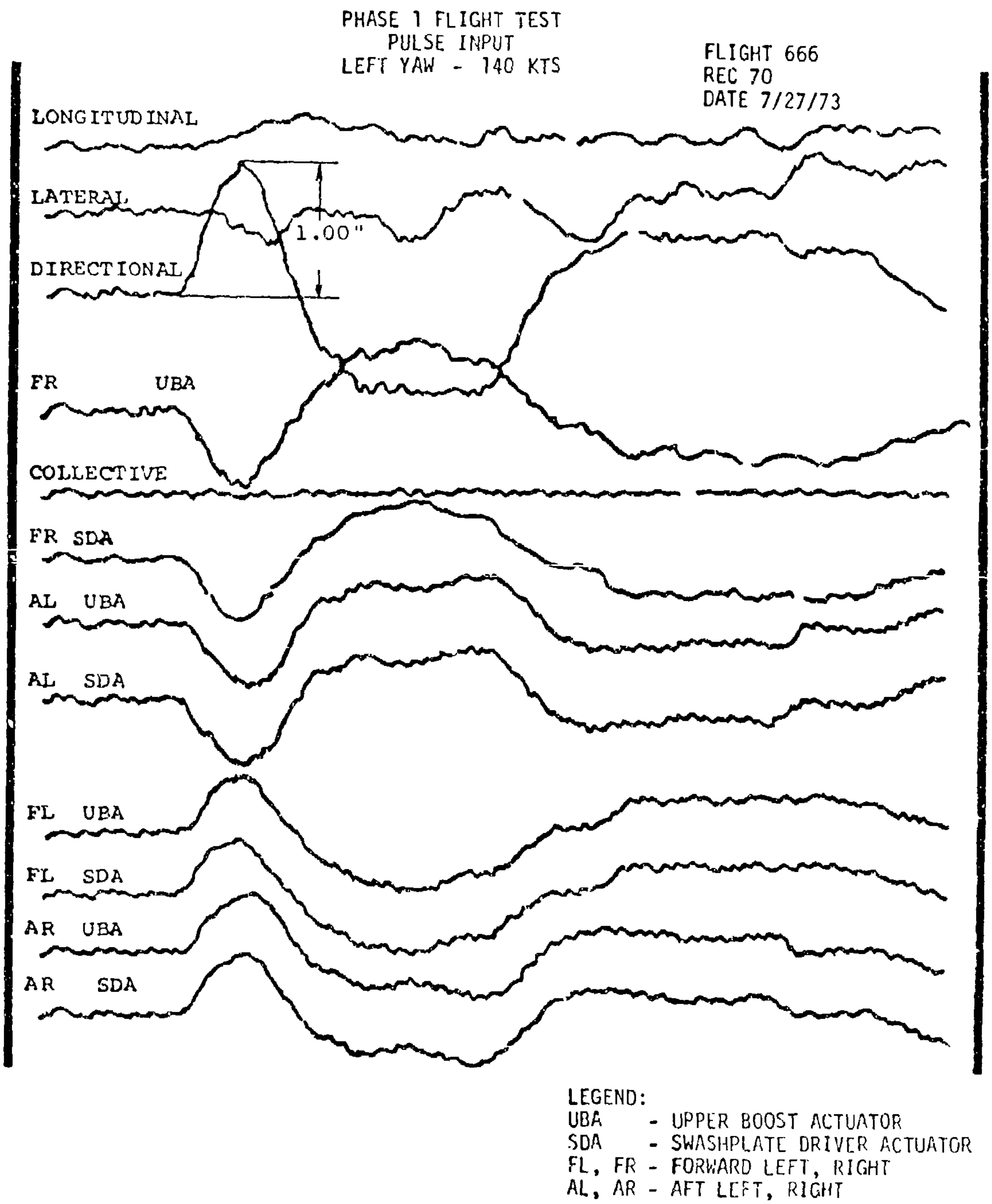


Figure 32. Typical Test Data, Comparison of Electrical With Mechanical System Outputs, Pulse Inputs.

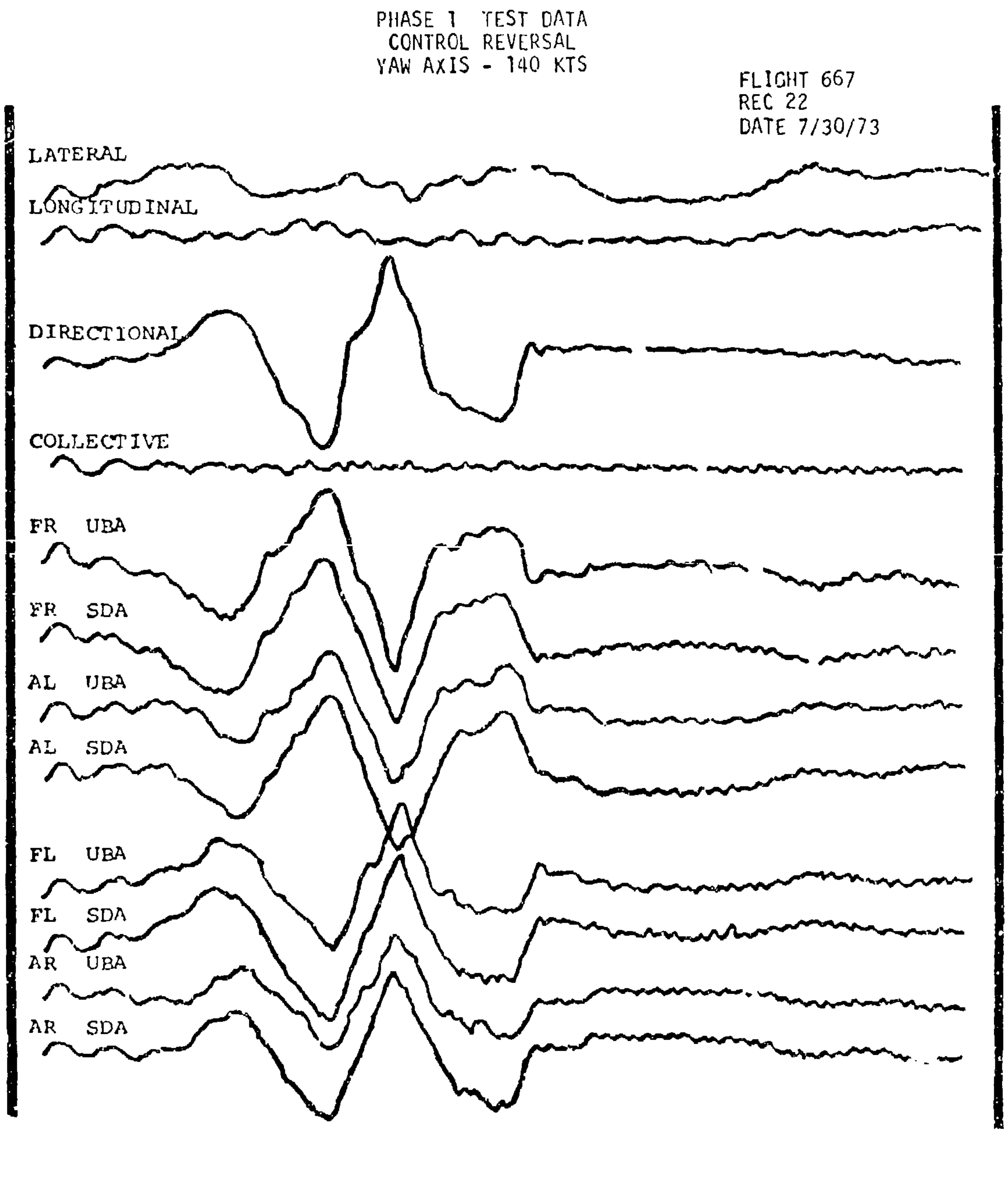


Figure 33. Typical Test Data, Comparison of Electrical With Mechanical System Outputs, Control Reversal.

Control Sensitivity

This test provided an early evaluation of DELS. Pilot's comments were:

1. "Rotor system and aircraft response was very tight with no detectable slop or dead band, with even the smallest increment of control input. There was no tendency to have PIO at any condition."

2. In ground borne test, pilot induced vertical and lateral-directional oscillations were well damped with no tendency to vertical bounce or mechanical instability.

3. In hovering flight, response to control was representative of a tight mechanical system. Pilot-induced sine-wave excitations in all axes revealed no tendency to any PIO.

   In forward flight, the pilot excited control frequency sweeps in all axes at 80 and 140 knots. Excitation was a sine wave form with $\pm$15- to 20-percent control authority and frequencies varying from 0.5 to 10 Hz. Control fidelity was true to the accuracy of the instrumentation. Typical data is shown in Figure 34. Rotor response was correct to the slightest DELS control input at any frequency. The (back driven) mechanical controls in the cockpit did not move with the low-amplitude, high-frequency excitations.

DELS First Failures

First failures were inserted into the number one DELS. Following each trial, the number two channel became active. Failure warning and status indications were all correct, and there were no detectable control disturbances following the failure or the reset to the number one DELS after removing the failure.

AFCS Interface Failures

Flight tests were preceded by a ground run to calibrate the AFCS simulator test set and to define authorities in each direction. The AFCS test set simulated AFCS signals. A photograph of the equipment is provided in Figure 35. The calibration confirmed that the dynamic test objectives could be met with the following authorities:

1. Longitudinal, up and down - 50 percent.

2. Lateral, left and right - 50 percent.

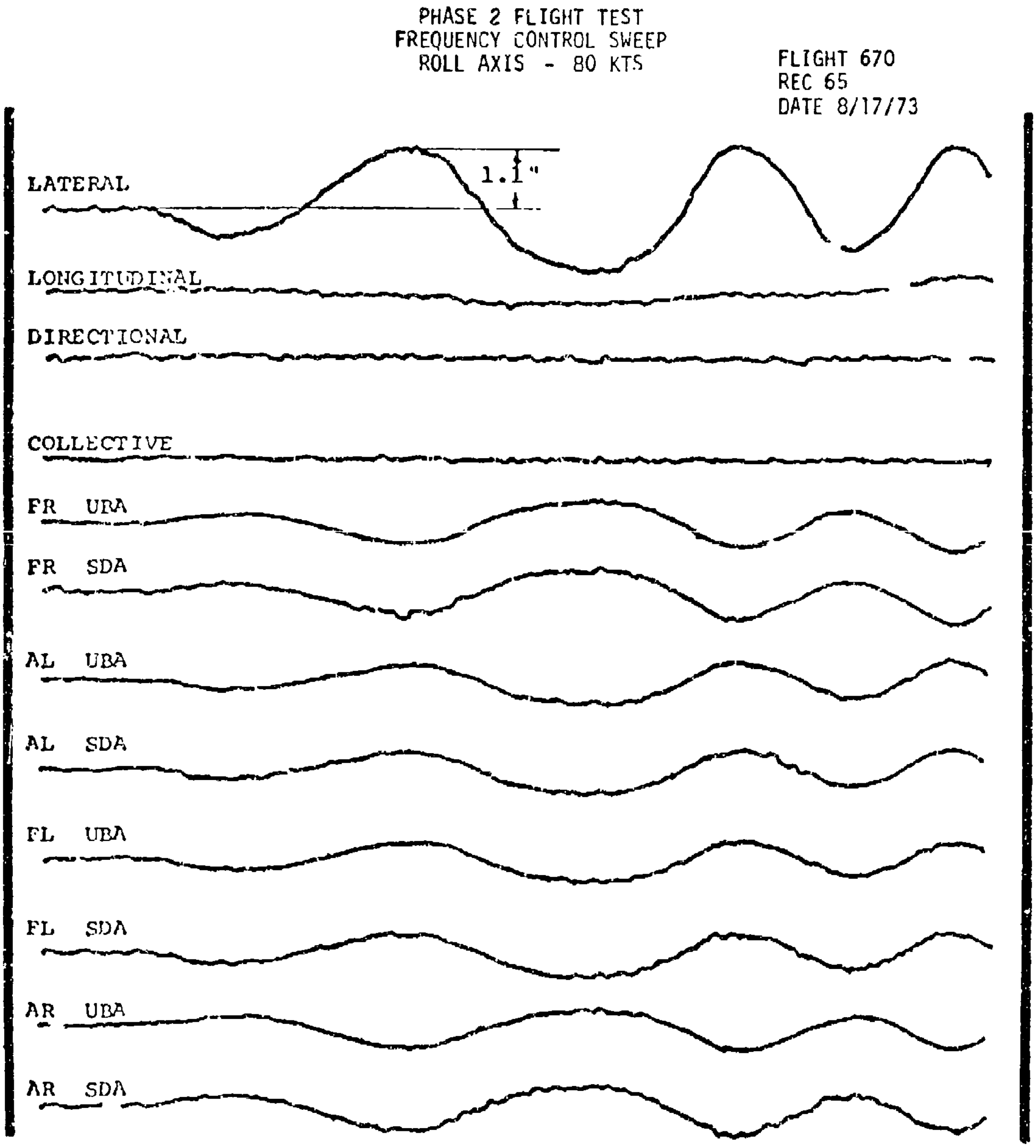


Figure 34. Phase II Typical Test Data, Control Sweep.

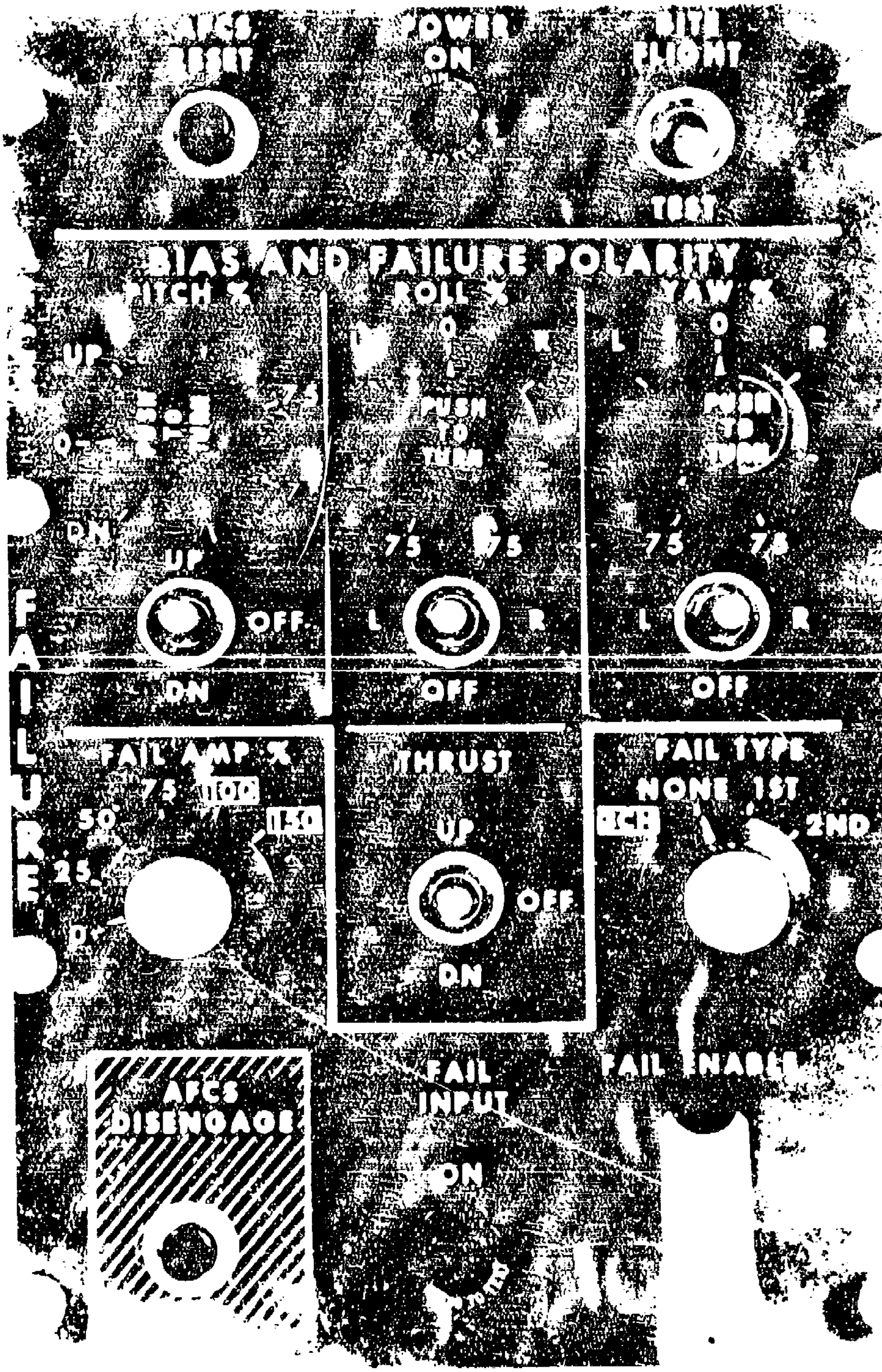


Figure 45. AFCS Simulator Test Set.

3. Directional, left and right - 75 percent.

4. Thrust (Vertical) up and down - 100 percent.

Inflight AFCS first and second failures were inserted. The DELS rejected (no response) each failure and provided the AFCS FAIL or AFCS OFF status display.

Triplicated (hard-over) AFCS failures were induced to evaluate transients, available recovery times, and control margins, and to substantiate the differential authorities intended by design. See AFCS Interface.

Note

The probability of triple failures of the actual AFCS is very low. However, the possibility of a conceptual error or maintenance error during the developmental program was further justification for hard-over tests. The dynamic hard-over failure tests were completed with acceptable results and $\geq$ 1-second time delay prior to pilot recovery. The reduced instrumentation data is given in Table 3.

The abrupt control step characteristic of the AFCS triple failures provides an unmistakable cue to the pilot for an immediate correction. Pilot corrections of .2 to .3 second to unannounced AFCS hard-overs would be expected when his hands are off the controls. In either case, the AFCS authorities for these axes are judged to be acceptable for recovery from a failure at any point of the flight envelope.

The evaluation of longitudinal,lateral and vertical control margins demonstrated acceptable margins and power with full authority in both directions.

The directional control margin evaluation revealed an unsatisfactory margin for a full-authority left differential input (1.5 inches) on the right pedal at 80 KIAS and above. The basic static pedal position gradient requires a .4-inch right pedal at 80 KIAS and a .6-inch right pedal at 120 KIAS and above. The control margin evaluation at 130 KIAS with a 1.5-inch left differential revealed that a .4-inch right pedal was available. In smooth air and level flight, control was marginally acceptable; however, in light turbulence or maneuvers, right pedal was frequently against the stop, and the response was very slow.

This deficiency did not affect the remainder of the Task II test program; however, correction was mandatory prior to Task III flight test.

TABLE 3. PHASE II AFCS HARD-OVER FAILURE TEST DATA

AXIS	DIRECTION	TIME* DELAY	ATTITUDE RATE AT RECOVERY	ATTITUDE AT RECOVERY
LONGITUDINAL	UP	1.3 SEC.	11.4°/SEC.	7.4°
	DOWN	1.4 SEC.	10.7°/SEC.	.8°
LATERAL	LEFT	1.7 SEC.	20.5°/SEC.	20°
	RIGHT	1.6 SEC.	25.2°/SEC.	21°
THRUST	UP	INDEFINITE	≈ 1000 FT/MIN	--
	DOWN	INDEFINITE	≈ 1000 FT/MIN	--
DIRECTIONAL	LEFT	1.3 SEC.	6.9°/SEC.	5.4
	RIGHT	1.1 SEC.	6.7°/SEC.	3.8
LEFT ROLL AND RIGHT YAW		1.0 SEC.	16.1/3.5°/SEC.	6.4°/1.9°
RIGHT ROLL AND LEFT YAW		1.2 SEC.	19.5/5.1°/SEC.	12.0°/5.1°

* Time delay prior to pilot recovery.

NOTE: Airspeed - 130 Knots.

Note

The deficiency was corrected prior to the AFCS flight tests. A one second or greater time delay was demonstrated. Reasonable conditions on recovery indicate the acceptability of the AFCS DELS interface features.

The step input of the hard-over failure provides an immediate cue to the pilot that requires no conscious analysis for correction. This cue is identical to a heavy gust upset, and the pilot will immediately apply a correction to maintain the desired flight condition.

Stress loads, as indicated on the Cruise Guide indicator and monitored by recorded instrumentation data, were well below endurance limits in all failure and recovery maneuvers.

DELS Single-Channel Flight

An evaluation of flight on each DELS channel as the only operating channel was conducted to ensure proper operation and performance of each.

DELS first and second failures were simulated by cycling hydraulic boost pressure off then on at 130 KIAS. Failure indications were correct in all configurations, and no detectable control transients were observed in either first or second failures or in resetting failed channels. Following second DELS failures, flight maneuvers representing those required to conduct an approach and landing were flown on each of the three DELS channels. The time spent on each channel was approximately six minutes. Each DELS channel performed properly in all respects.

The DELS met all requirements in Phase II and was judged acceptable in all respects to proceed to Phase III.

Phase III - Pure Fly-By-Wire

In Phase III, the helicopter was tested in standard flight conditions (Table 2). All test conditions were handled satisfactorily, and DELS performance was correct in every respect. Preflight and hover tests were completed satisfactorily, and the aircraft was accelerated to 80 KIAS in a climb for data collection. At 80, 120 and 140 KIAS at test altitude, 3,000 H_d, control response, sensitivity, and stability were evaluated in pilot-induced variable-frequency sweeps ($\pm$10 percent control from .5 CPS to 10 to 15 CPS in all axes). The rotor response was that of a tight mechanical system with no tendancy to PIO, and the DELS actuator outputs exhibited positive damping with no oscillation tendancies.

An airspeed sweep from 0 to 150 KIAS was conducted.

Use of the Built-In Test Equipment

The BITE test was performed each day prior to flight. It was not performed prior to each flight when more than one flight was made a day.

Sequencing through BITE required approximately one and one-half minutes. There were twenty-six automatic tests for each channel. The BITE sequence was followed by boxing the controls while monitoring the failure status panel. These procedures constituted a daily inspection of DELS. This daily inspection can be performed in 5 minutes.

DELS Failure and Malfunction Record

No inflight malfunction or failure occurred (induced failures were used to test channel switching).

The failures that did occur were during the early integration testing and ground tests. These are enumerated below.

1. Failure at extreme longitudinal control position; tripped longitudinal fault check. Found failed Zenier diode resulting in mistrack of active model stick mixer (one time).

2. Failure and shutdown of actuators occurred at extreme thrust position; failure sensitive to input velocity. Failure attributed to contamination of uncoated circuit board causing mistrack of active and model stick mixer (one time).

3. Unit failed BITE test. Found open pin on bit card (one time).

4. At end of BITE test, unit was found to have failed. Found mistake in wiring on BITE card. Mistake occurred during modification program (one time).

5. Active light emitting diode failed to light. Found pushed back pin in control unit. Another pin found in junction boxes.

6. Servo loop oscillates when low side of torque motor is grounded. Loop gain goes high and maximum spool position attempts to shutoff, but 20 Hz oscillation will not allow shutoff. Shutoff occurs with velocity command input, e.g., pedal movement. Fix added

latch on spool position maximum. Problem represented design problem which was corrected.

7. Servo-amplifier comparator trips at limit actuator position. Found active and model amplifiers do not track at saturation point. (Work around during Task II flights and instituted design modification during Task III layup.)

8. System failed BITE track test; channel 3 DELCU replaced. Unit checked out satisfactorily at General Electric. Transducers were renulled — no further problem.

9. Pilot station display dimming failed. Found failed Zenier diode in dimming circuit. Replaced with higher rated unit.

10. Two Light-Emitting Diodes (LED) failed in DEL status panel. Replaced LEDs; suspect mechanical damage in one case. Added protection for LEDs to face of panel. The second LED appeared to be a plain failure.

11. S/N 8 driver actuator found leaking at General Electric before delivery. Replaced piston rod seals. Found improper finish on rod, reworked piston rods.

12. S/N 6 driver actuator removed from aircraft for static leakage - 1 drop/minute, returned to Bertea Corp.

Note

During the AFCS tests, reported in Volume III, there were no DELS failures.

Breakdown of Flight and Component Operation Hours

Total DELS Flight Time

DELS Flight Test (this volume)	18:49
Pendulum Absorber Flights (flown prior to layup for AFCS)	9:27
AFCS Flight Test (Volume III)	232:10
Post ATC Demonstrations at Ft. Eustis, Ft. Rucker, Ft. Belvoir	54:34
Total	315

Pure Fly-By-Wire Time

DELS Tests		8:02
AFCS Tests		266:30
	Total	274:32

Operation Time on DELS Control Units

The operating time on the DELS control unit is indicated in Table 4.

TABLE 4. DELS CONTROL UNIT OPERATING HOURS

S/N	INTEGRATION TEST TIME (HRS.)	AIRCRAFT* TIME (HRS.)	TOTAL* TIME (HRS.)
73001	198	47	245
73002	236	47	283
73003	228	53	281
73004	69	1,917	1,986
73005	68	79	147
73006	67	1,890	1,957
73007	(Qualification Test Unit)		
73008	69	286	355
73009	28	1,969	1,997
TOTALS	963	6,288	7,251
DELS Time during AFCS Integration Test	271	--	271
TOTALS	1,234	6,288	7,522

*Includes AFCS flight tests.

Follow-on DELS Performance

After the completion of the DELS flight test program, the 347 helicopter was used to test pendulum absorbers. The mechanical backup of the Phase II system configuration was used. Flight time was nine hours and twenty-seven minutes. DELS functioned perfectly.

The DELS equipment functioned perfectly during the AFCS testing (Volume III) and the post test demonstrations; 232 and 55 flight hours, respectively. The AFCS tests proceeded in a progressive manner where each step represented a "different" AFCS system. In many instances, the AFCS initial conditions, the subsequent experimentation, or the outright failures resulted in erratic and very unsatisfactory performance. The DELS itself was unaffected and continued to perform perfectly. Flight safety was never jeopardized by AFCS status or performance. The progressive development of AFCS in flight test was:

1. Basic Stability and Control Augmentation (SCAS).
2. Basic SCAS plus Altitude Hold.
3. Hover Hold and Load Controlling Crewman Controller.
4. Precision Hover Hold (Precision Hover Sensor).
5. Basic SCAS plus external load.
6. Basic SCAS plus Load Stabilization.
7. Hover Hold and Load Stabilization.
8. Precision Hover Hold plus Load Stabilization.
9. Precision Hover Hold plus Load Position Hold.
10. Automatic Approach to Hover.

Reliability Evaluation

Analytical reliability evaluation of the DELS design indicated a channel MTBF of 1,592 hours.

The HLH-347 demonstrator flew a total of 315 hours with no inflight failures of the three DELS channels, for a total of 945 channel hours. The normal method of determining reliability failure rates is the "point estimate" method (number of failures divided by total hours). In the special case of zero

failures, this results in an unrealistically optimistic failure rate (0 ÷ 945 = 0 failures per hour). An alternative approach is to assume that if we had operated a fraction of a second longer (e.g., 945.00001 hours) we would have had a failure and work the point estimate based on one failure (1 ÷ 945 = .0010582 failures/hour). This alternative is unrealistically conservative since, in fact, we did not have one failure. Clearly, however, the true failure rate lies somewhere between 0 and .001058 which indicates that the MTBF (1/failure rate) lies somewhere between 945 hours and infinity. A rigorous mathematical analysis will show that the "maximum likelihood estimator" of the true failure rate is achieved when the number of failures is assumed to be in the range of "minus ln .75" (.287682). Thus; the most likely failure rate = .287682/945 = .000304, with a corresponding MTBF of 3,284 hours. Note that .000304 falls in the range 0 to .001582 and that 3,284 falls in the range 945 to infinity.

Using the "most likely" MTBF of 3284, the predicted flight safety reliability for the channels as a system is .9(9)774 for a 2-hour mission. The predicted mission number one (abort) reliability is .9(5)888 for a 2-hour mission. The predicted maintenance malfunction reliability is .99817 for a 2-hour mission.

Since a failure rate is not normally considered well defined until the test time exceeds the MTBF, it is not justified to replace the analytical predicted MTBF of 1,592 hours with a 3,284-hour value. If the true MTBF is 1,592 hours, the probability of getting zero failures in 945 hours is 0.55.

CONCLUSIONS - PRIMARY FLIGHT CONTROL SYSTEM

Overall, the feasibility of a fly-by-wire control system was clearly demonstrated. Furthermore, pilots readily accepted fly-by-wire control. Particular conclusions include:

> The separation of primary flight control from stability and control augmentation proved beneficial. AFCS problems did not affect the integrity of the DELS.
>
> The function of the AFCS interface circuits, to limit any possible transients to a safe level, was positively established.
>
> The use of BITE was effective. There were no inflight failures of DELS. All airborne failures were deliberately induced as part of the test plan.
>
> Fly-by-wire was equivalent to the mechanical system (input to output relationships).
>
> Handling qualities were not changed significantly. One pilot's opinion was that "it flies like a new, well-adjusted mechanical system". The handling qualities will not deteriorate with time because there are no mechanical joints to wear or be adjusted. It was shown in the test that hysteresis was less in the fly-by-wire system than in the mechanical system.

The successes of this program were obtained despite the constraints of the program, which included budgetary limitations and the test vehicle used. Some of the effects of these constraints were:

> A driver actuator was used instead of a swashplate servo-actuator to avoid the cost of major structural and skin line modifications in the rotor pylon area.
>
> Electrical and hydraulic prime power sources were the original CH-47C systems with add-ons to facilitate the triple redundancy of the DELS.
>
> The cockpit controls were the original CH-47 components, and stick position transducers were installed in the cabin heater closet. These expediencies prevented major redesign of the cockpit area and the nose of the aircraft.
>
> Components just met the minimum requirements for flight clearance. The system power supplies, wiring, transducers, activators, and black boxes were not hardened for lightning strike or nuclear radiation.

PART 2 - COCKPIT CONTROLS SUBSYSTEM (CCS)

The contract statement of work for Heavy Lift Helicopter Advanced Technology Component Program fly-by-wire flight control program included: trade studies to select an HLH controller configuration compatible with fly-by-wire design, fabrication of an experimental controller subsystem; and laboratory evaluation and test of the experimental model. The following reports the program results.

TRADE STUDY

Required trade studies were performed by a subcontractor, CAE Electronics Ltd., Montreal, Canada. Principle requirements and constraints imposed on the study were:

1. CCS shall accommodate pilot controllability of a direct electrical linkage and specifically, shall provide the pilot a satisfactory level of controllability upon unexpected loss of stability and control augmentation.

2. CCS shall accommodate parallel drive; i.e., AFCS "autopilot type" inputs.

3. Synchronization shall be provided between pilot and copilot controls.

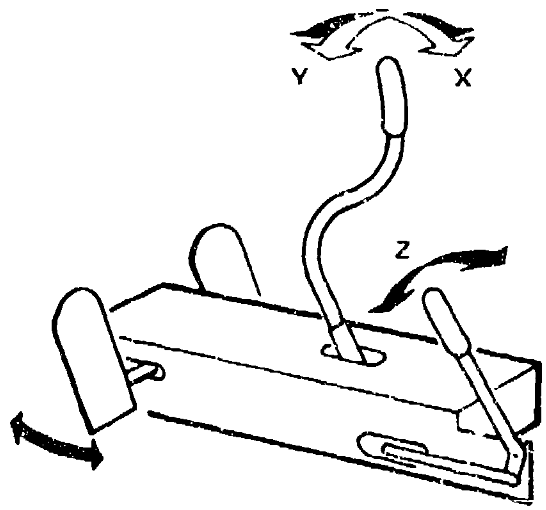
4. Variable force feel shall be included to provide maneuver cueing and/or envelope limiting.

5. Redundancy shall be appropriate to the HLH fly-by-wire flight safety and vulnerability requirements. Driver actuators and transducers shall be at least dual.

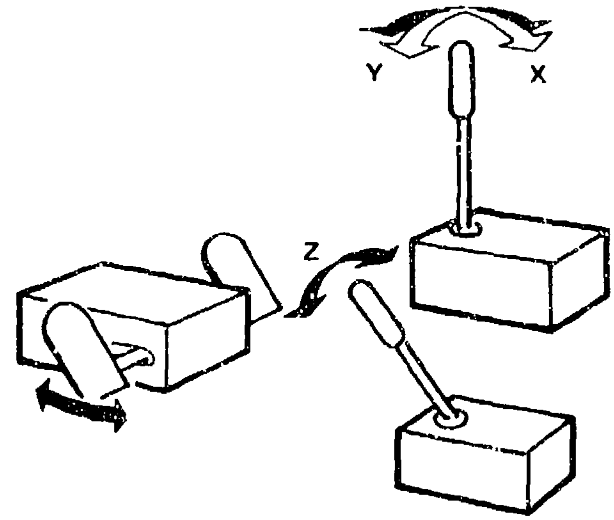
6. CCS must be compatible with crash survivable seats.

Configurations for study ranged from four-axis sidearm controllers to improved versions of conventional controls wih electrical synchronization. The configurations were all of the 2+1+1 type, meaning that the longitudinal and lateral motions are controlled with one control, the collective on another, and the direction with pedals. Figure 36 shows a sketch of the 2+1+1 schemes that were considered.

Other candidates, such as the four-axis and 3+1 systems, were not considered because of their anthropomorphic problems.



2 + 1 + 1 CONTROLLER CONFIGURATION A



2 + 1 + 1 CONTROLLER CONFIGURATION B

Figure [illegible]5. Sketch of Trade Study Recommended Improved Electrical Controllers.

Also, a packaging that would satisfy all functions at the required redundancy level appeared prohibitive.

A critical-item development specification for the recommended 2+1+1, electrically synchronized system was established. Proposals were solicited from industry for the system, and three companies replied. An evaluation of the proposals revealed that the electrically synchronized system was too complex and too costly. With Army concurrence, it was decided that a mechanically synchronized system with separate, programmable force-feel/cockpit controller-driver actuators was the better approach for the HLH fly-by-wire program. The following paragraphs describe the resulting ATC program.

DESCRIPTION

It is convenient to describe the mechanically synchronized CCS in terms of the control axis and the electronics that are attached. The major components are the longitudinal/lateral control, the collective and the directional. Associated with each axis is a programmable force-feel unit/cockpit controller-driver actuator system (PFFU/CCDA) and stick position transducers. In addition, toe-operated brakes are provided on the directional pedals.

GEN[illegible] DESIGN REQUIREMENTS, MECHANICAL PARTS

[illegible] requirements that apply to the mechanical design are:

1. The CCS shall accommodate the 5th through 95th percentile Army aviator.
2. The synchronization load path shall ensure that neither pilot will lose control as a result of a single 12.7mm projectile.
3. The basic synchronization linkage must be protected against jams by shear sections or equivalent devices.

Mechanical Design

The test stand for this equipment is shown in Figure 37.

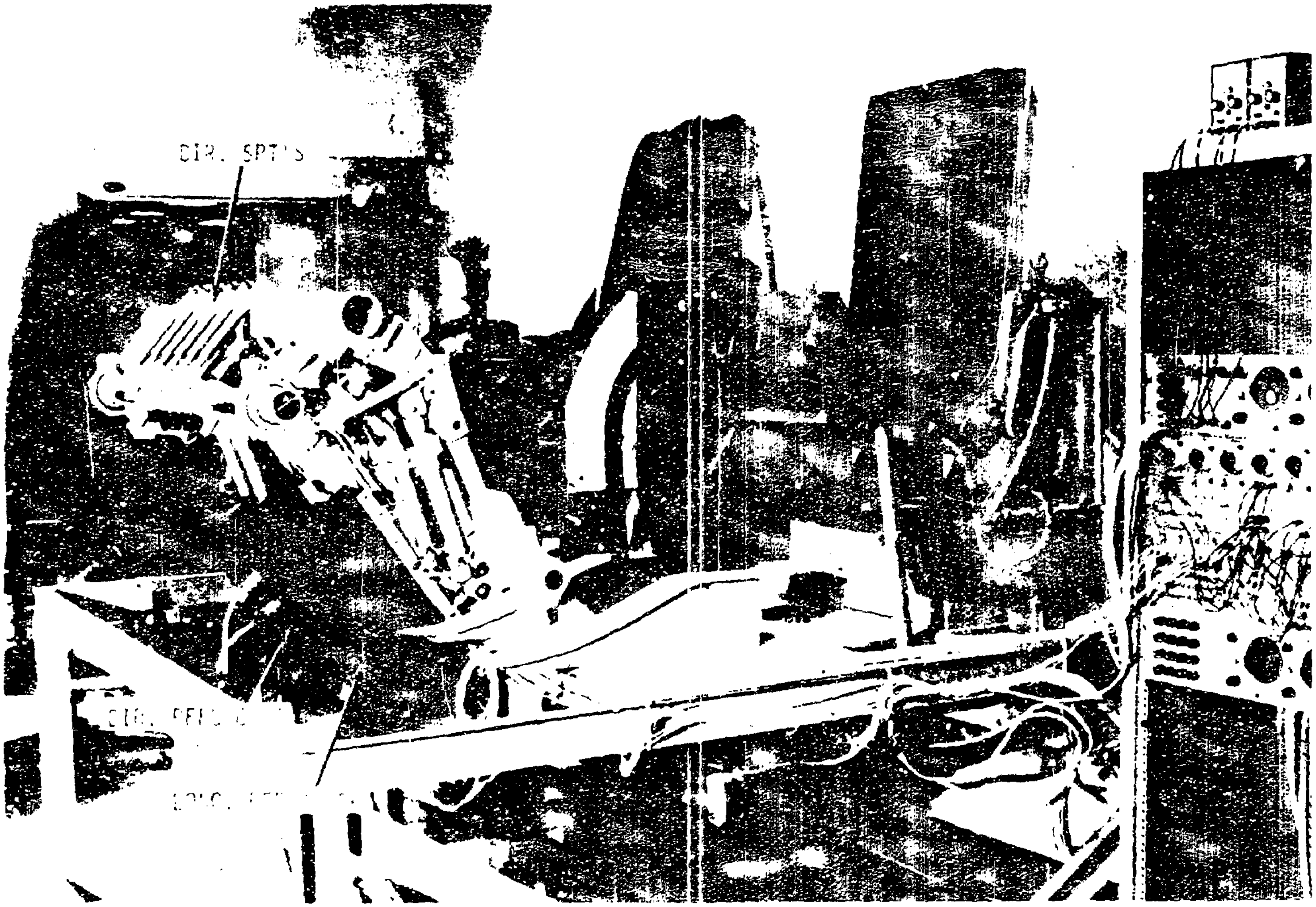


Figure 37. Cockpit Controller Test Stand.

The general design features are:

1. Torque tubes (and push-pull tubes) are two inches in diameter to survive a 12.7mm projectile.

2. There are dual load paths at critical pin joints.

3. Shear pins are used to attach SPTs and PFFU/CCDA lever arms to torque tubes. All shear pins are identical and easily replaceable.

4. There are counterbalances on pitch and thrust (longitudinal and collective). Roll (lateral) balance is in the form of a reverse image (mass). The suspended rudder pedal design gives the yaw (directional) inherent balance.

5. Rolling element bearings are used to minimize friction.

Longitudinal

The longitudinal/lateral cyclic stick can be seen in Figure 37. The pivot joint for the longitudinal torque tube (below the cockpit floor) can be seen in Figure 38. The torque tube extends from copilot to pilot controls. It is attached to hard structure at both ends and in the center.

The longitudinal (pitch) stick travel is ±5.5 inches at the grip. Adjustable stops (Figure 38) are provided for rigging. The SPT stroke is ±1 inch.

The PFFU/CCDA push rod is attached to a lever arm off the torque tube. This lever arm, like the six for the SPTs is attached to the torque tube with shear pins. Typical shear pin installation can be seen in Figure 42. The counterbalance weight is identified in Figure 39.

Lateral

The lateral pivot joint is shown in Figures 38 and 39. Rotation around the pivot drives the lateral (roll) push-pull tube. This tube synchronizes the pilot and copilot sticks and drives the SPTs. See Figure 39. A rocker arm driven at one end of the push-pull tube connects to the PFFU/CCDA shaft. See Figure 40.

Lateral stick travel is ±4 inches at the grip. The SPTs stroke is ±1 inch. One side of the adjustable lateral stops is shown in Figure 38.

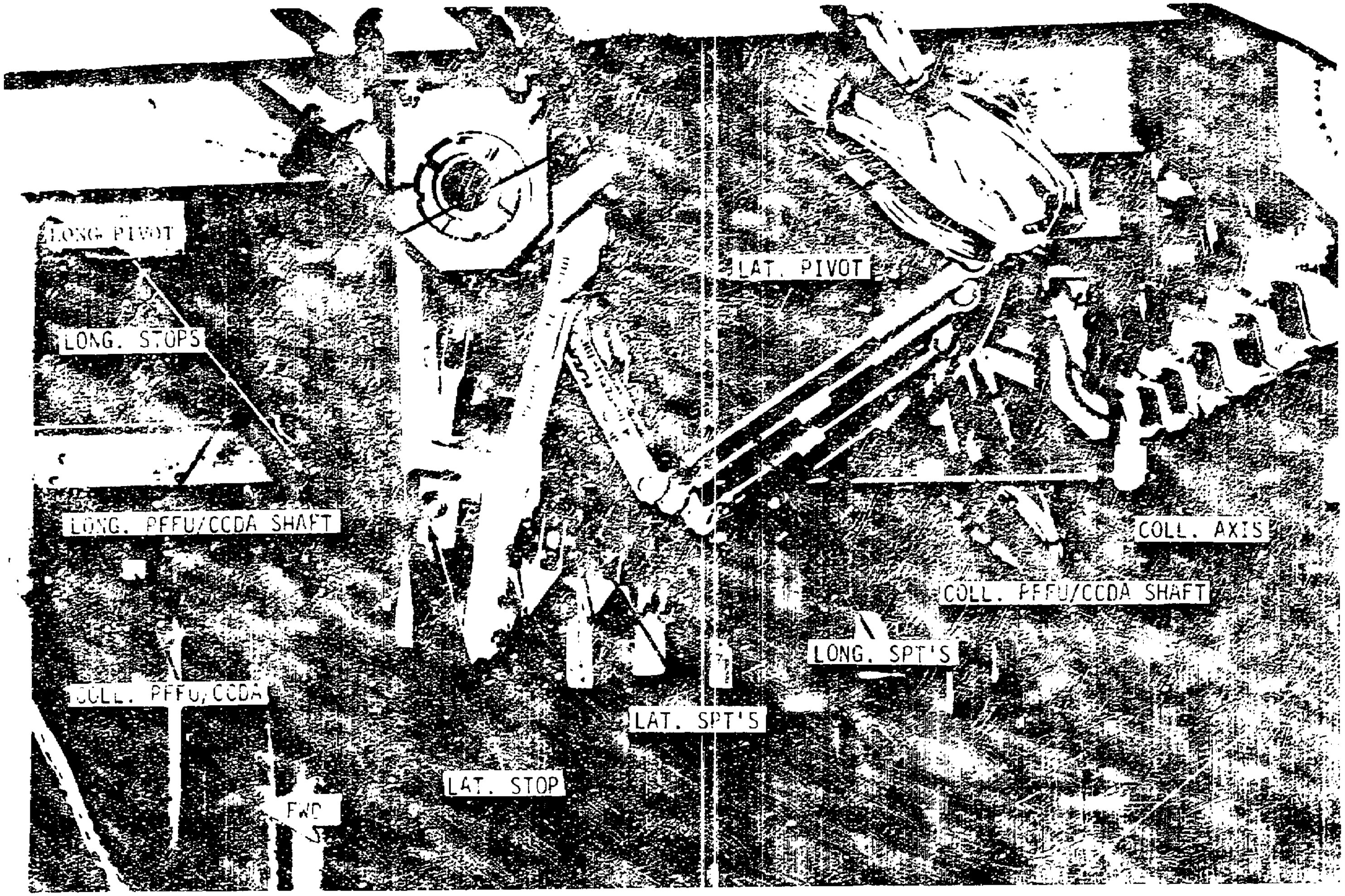


Figure 38. Longitudinal/Lateral Control Assembly Below the Floor.

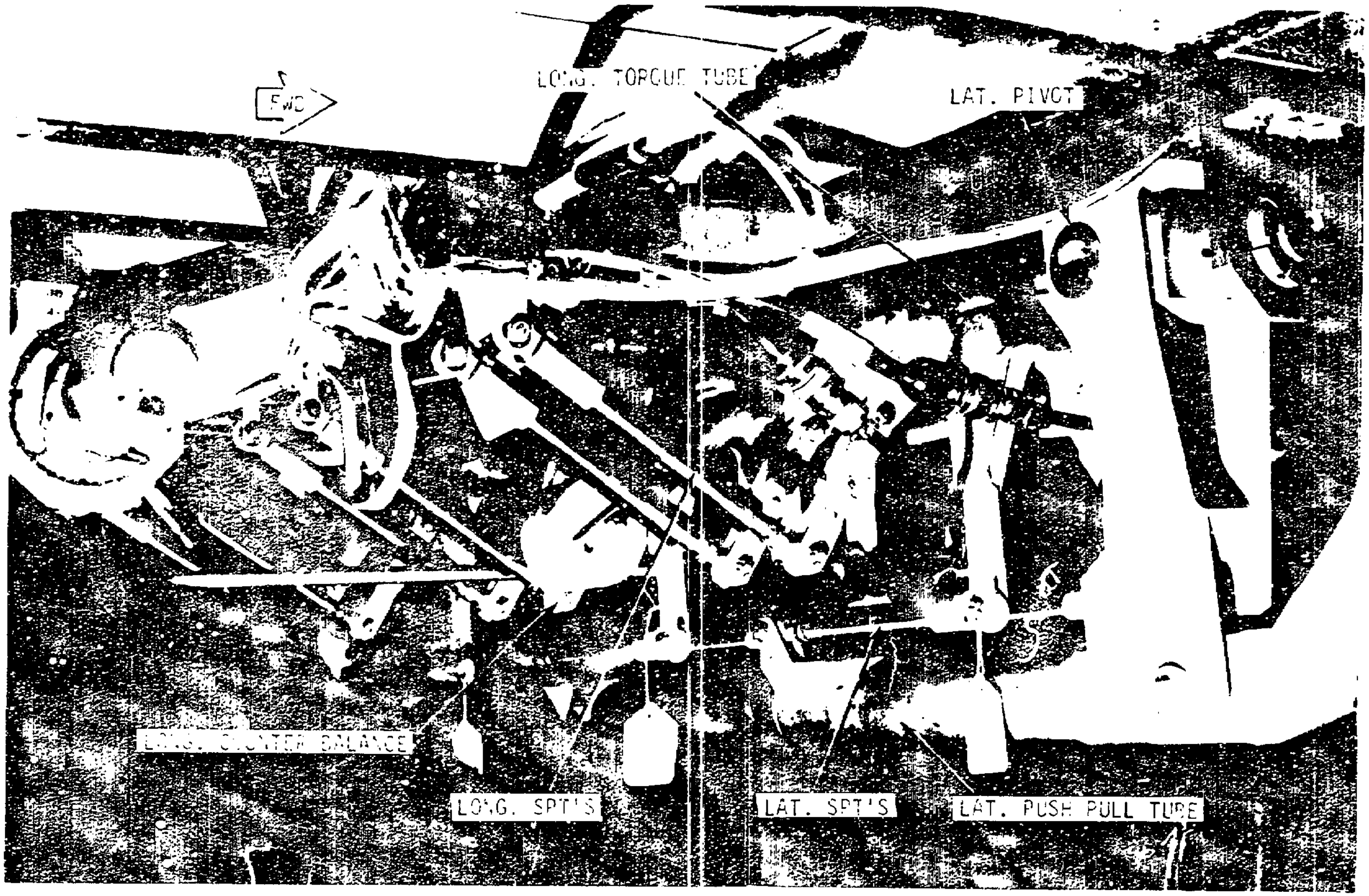


Figure 39. Longitudinal/Lateral Control Assembly Below the Floor.

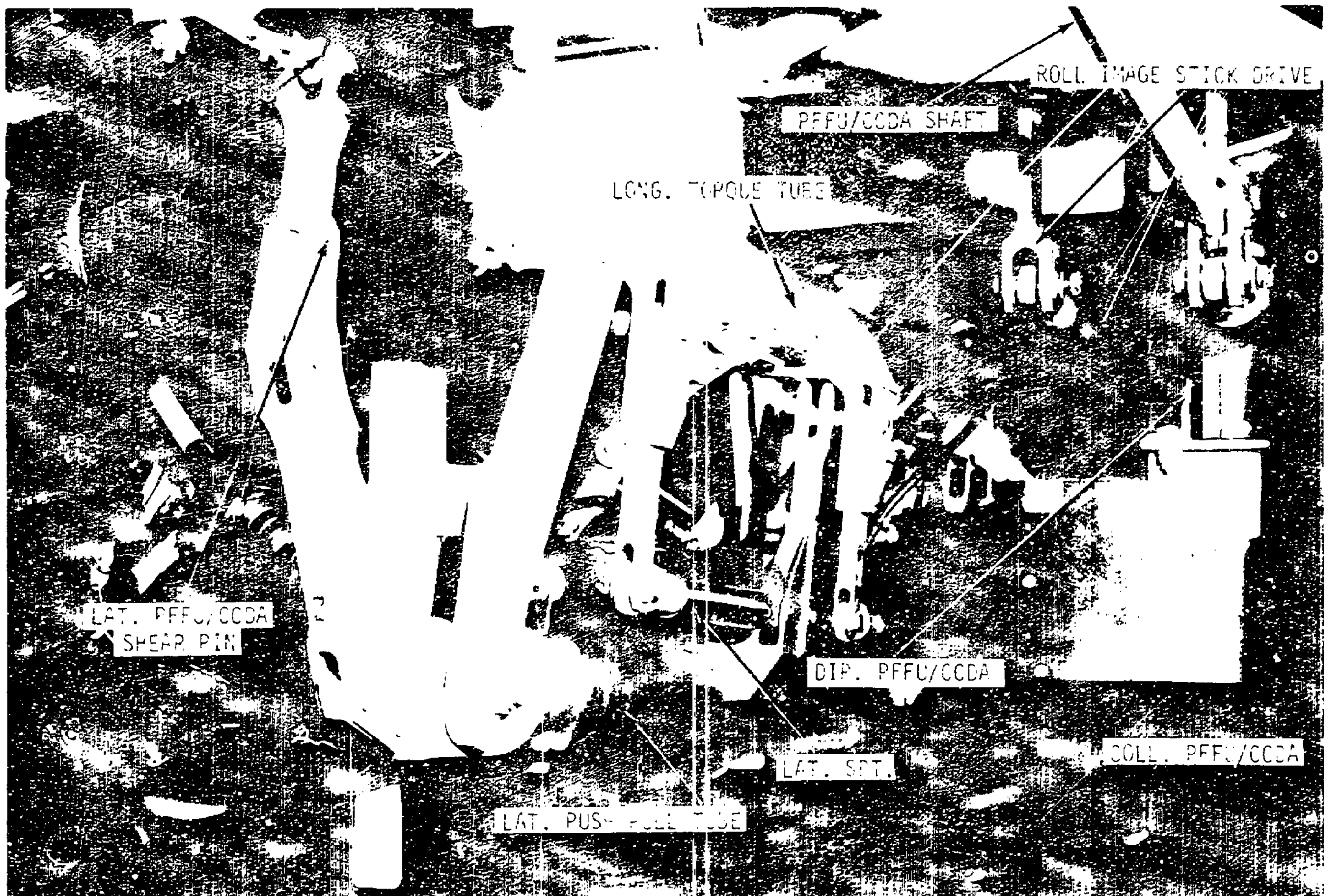


Figure 40. Lateral Controls Assembly Below the Floor.

Collective

Moving the collective lever (which can be seen in Figure 37) produces rotation of the collective torque tube, Figure 41. The torque tube assembly with SPTs, counterbalance, etc., can be seen.

The travel of the collective lever is ± 4 inches at the grip. Motion follows an arc of 11-inch radius. Stick position transducer stroke is ± 1 inch.

Directional

An overall view of the suspended rudder pedals is given in Figure 37. SPT installation detail can be seen in Figure 43. This photo also shows the SPT null adjustment, lock, and the redundant support, which allow operation with loss of one main support. There is clearance between the torque tube and the support.

Details of the rudder pedals and heel slides are presented in Figure 44. Pedal stroke is ± 2.5 inches. Pedal adjustment for pilot reach is ± 3 inches. The two sides, pilot and copilot, adjust separately.

The toe brake spring capsule (interface equipment) is shown in Figure 44. Transducers for brake-by-wire (not shown) will be installed parallel to the spring capsules.

DESCRIPTION OF PROGRAMMABLE FORCE-FEEL UNIT/COCKPIT CONTROLLER DRIVER ACTUATOR (PFFU/CCDA)

The Advanced Technology Component Program required the development of an integrated programmable force-feel unit and cockpit controller driver actuators. The necessary hardware was fabricated and evaluated statically in the CCS integration test stand.

PFFU/CCDA Functions, General

A simplified schematic of a PFFU/CCDA is provided in Figure 45. The functional descriptions that follow apply to all four axes unless otherwise noted.

Each actuator unit is mechanically connected to its respective control axis torque tube or push-pull tube. The PFFU/CCDA units can both react the pilot's control input motions or can backdrive the cockpit controls as functions of AFCS inputs.

The basic forces imparted to the pilot's controls are proportional to the displacement of the control form trim and the

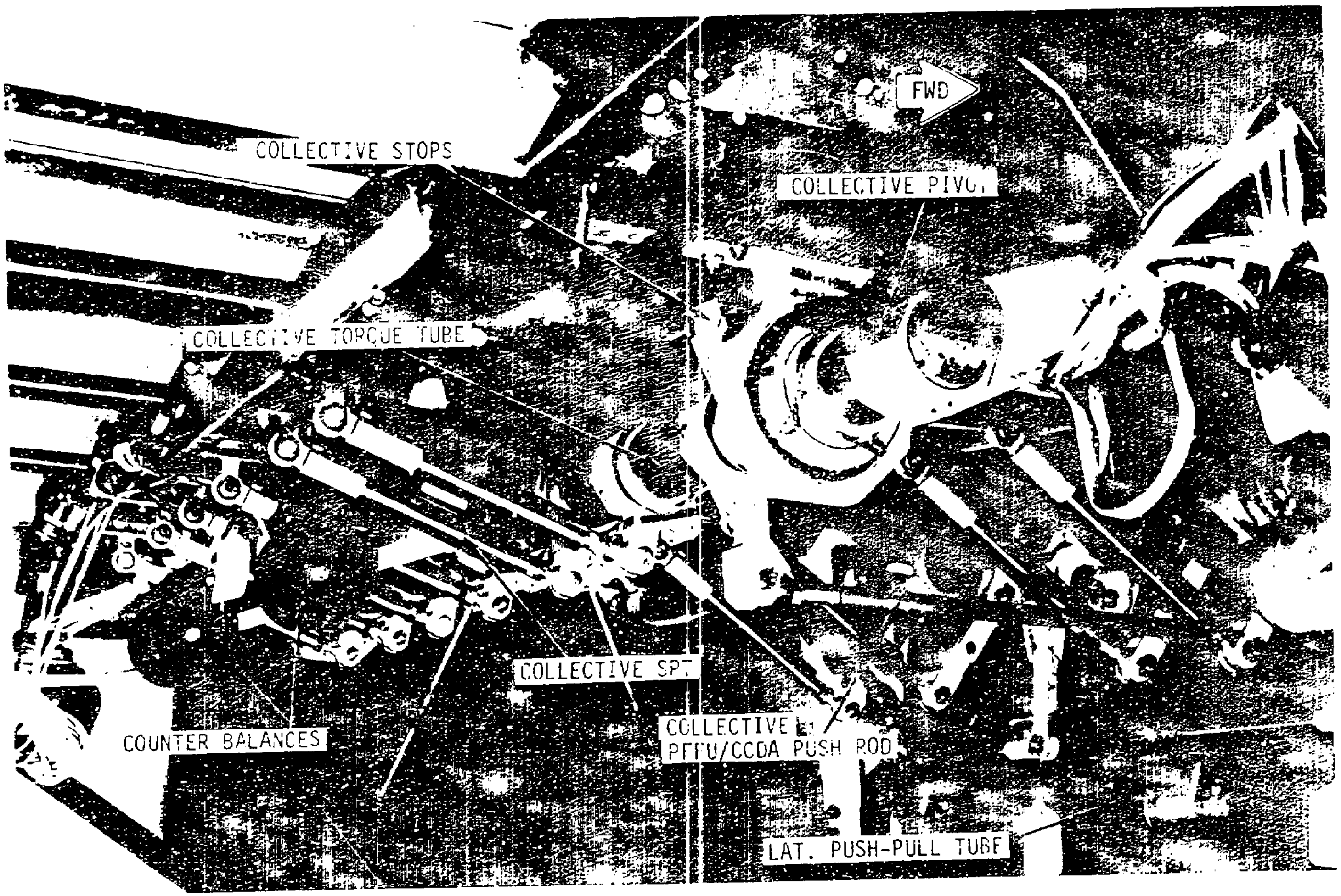


Figure 41. Collective Controls Assembly Below the Floor, Overall Rear View.

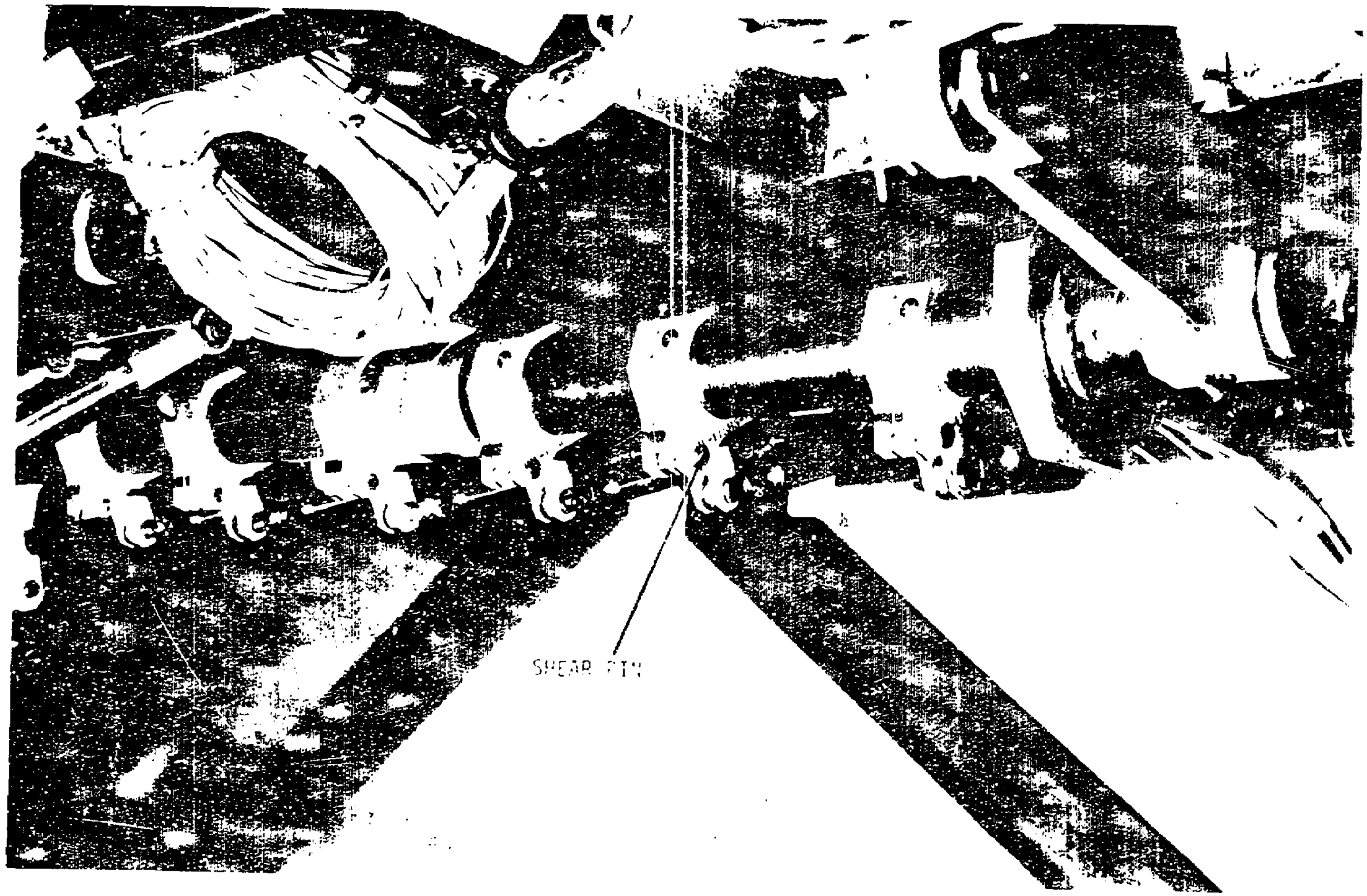


Figure 42. Collective Controls SPT Shear Pins.

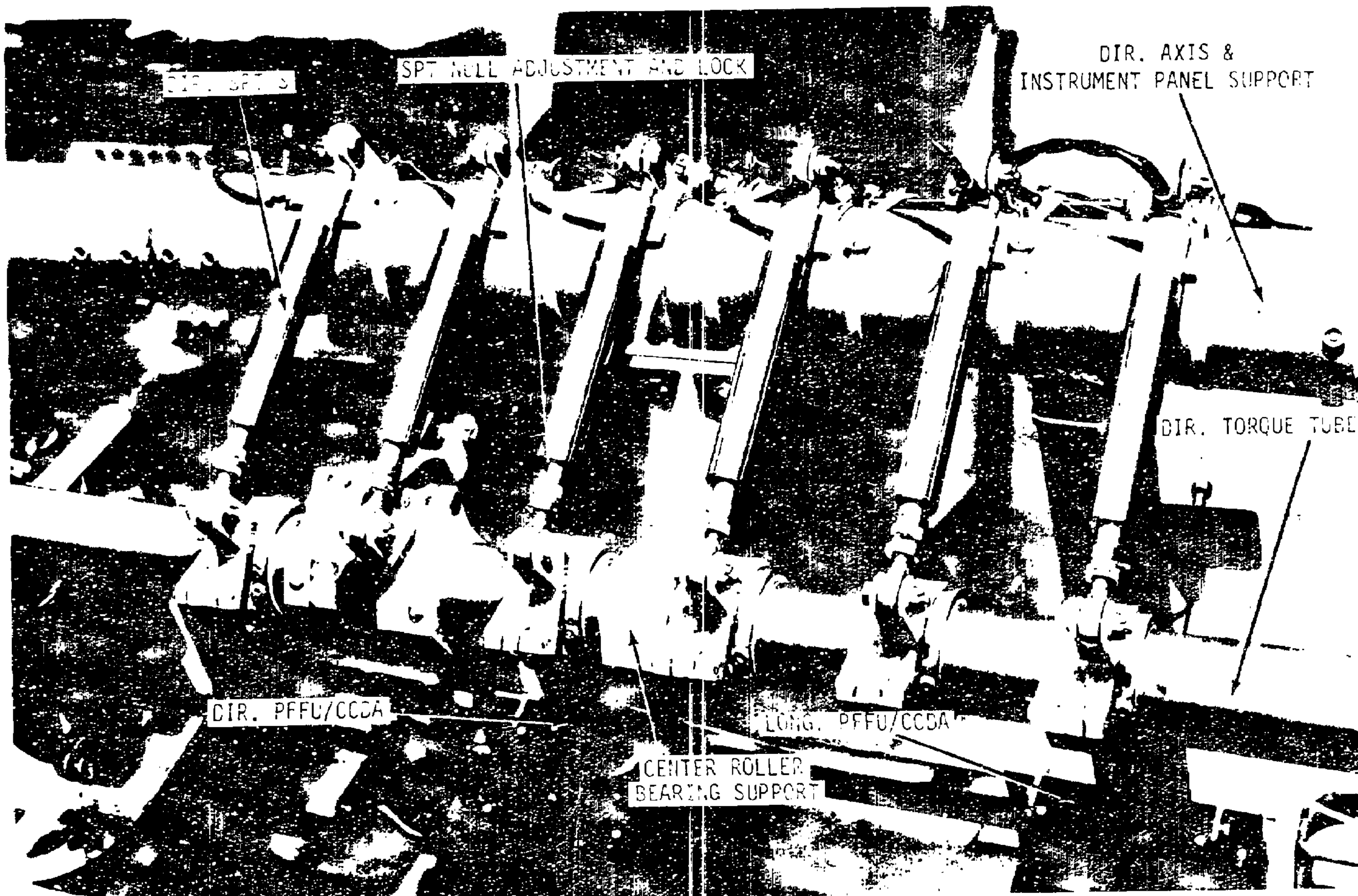


Figure 43. Directional Controls SPTs.

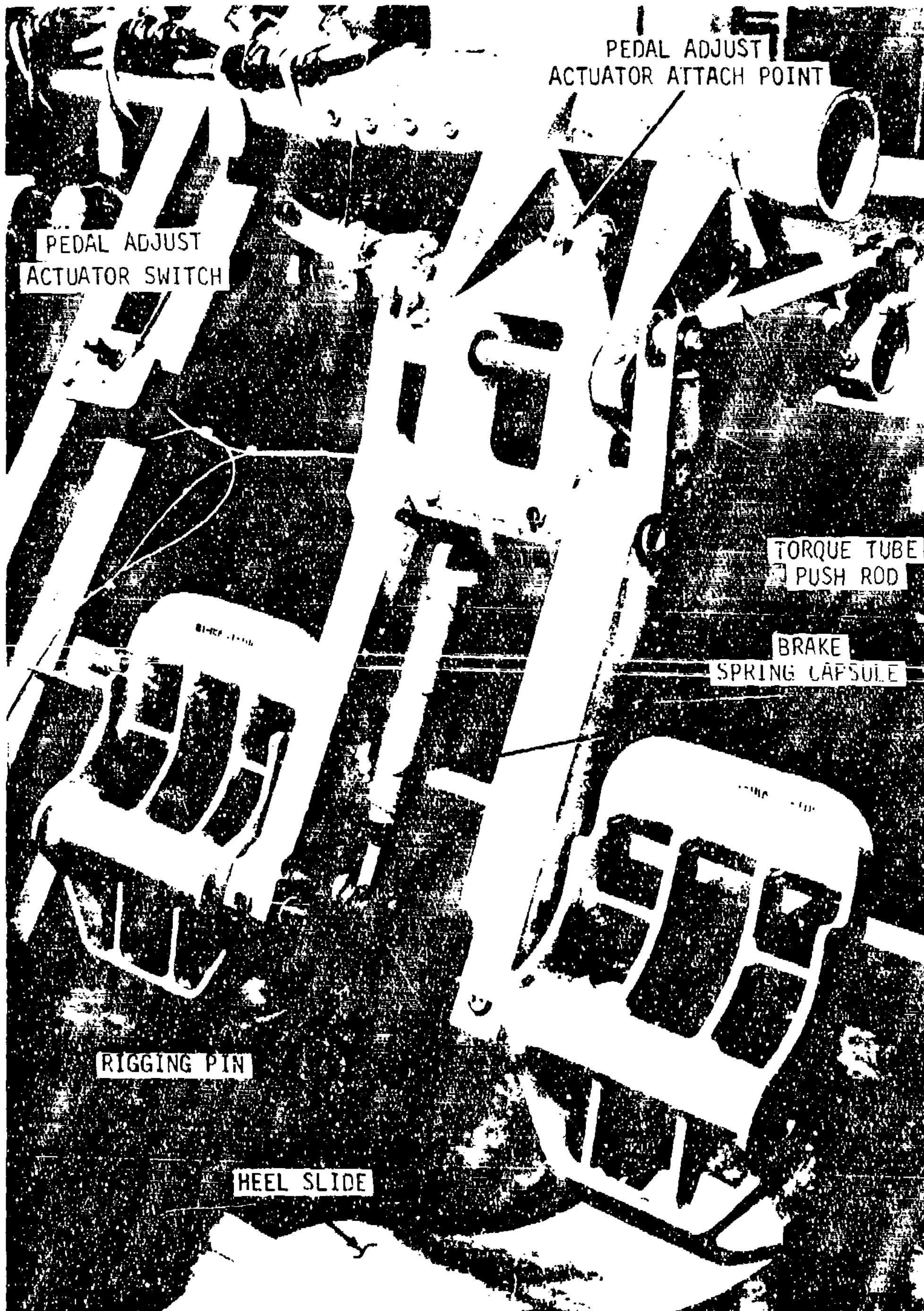


Figure 44. Directional Pedals Adjust and Heel Slides.

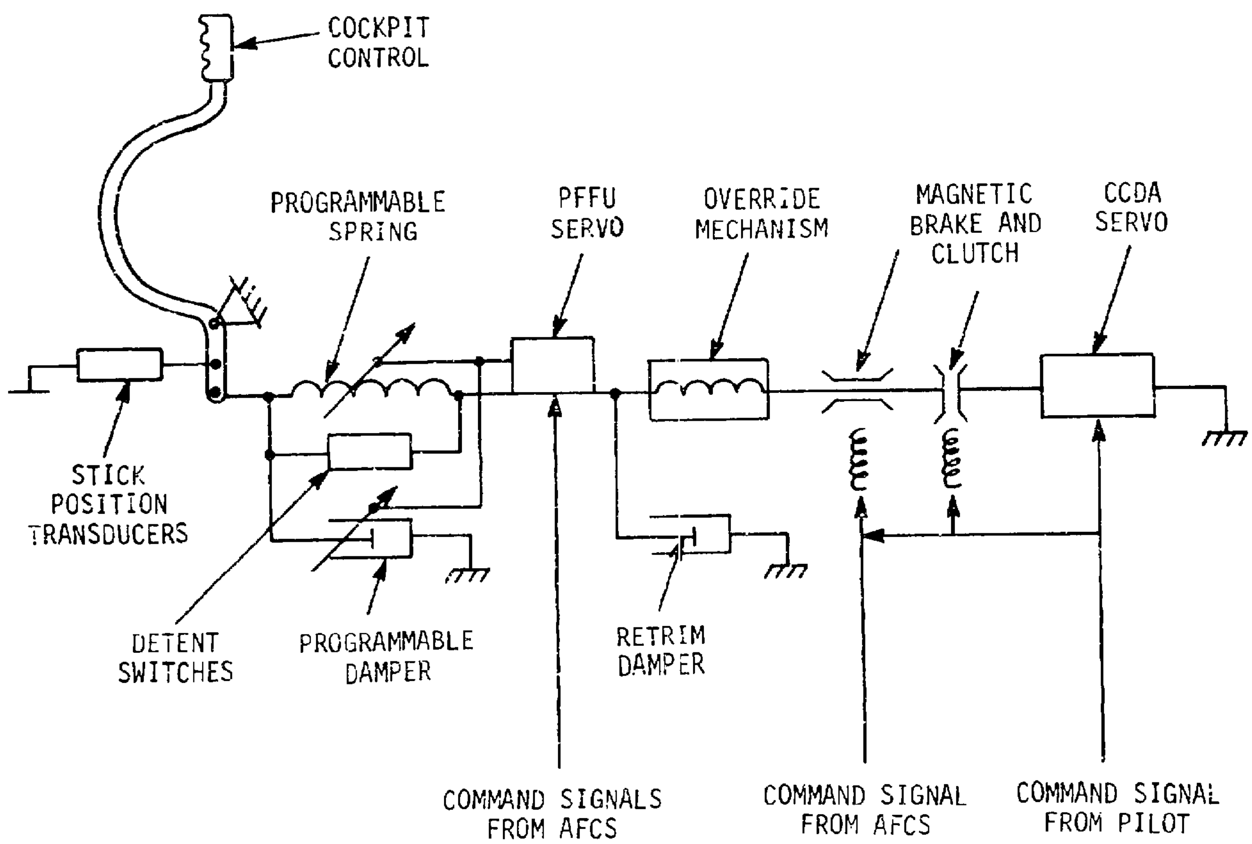


Figure 45. PFFU/CCDA, Simplified Diagram.

rate of displacement. The ratio of force to displacement is controllable in the PFFU in accordance with signals generated in the AFCS. Damping coefficient is also variable.

Note

The AFCS control laws to vary force feel and damping are not established. Flight evaluation is required. Possible parameters are engine torque, pitch attitude, bank angle, airspeed and altitude rate.

When the programmable force feel is not in use for any reason, a fixed force feel and damping is available.

The CCDA receives positional signals from the AFCS in certain functional modes. The interface signals are given in the block diagram of Figure 46. For example, automatic approach to hover and, when the load controlling crewman is flying the aircraft. In this way, AFCS low-frequency command maneuvers of the aircraft are reflected at the pilot's controls. This results in transient-free mode switching and AFCS disengagement, and minimizes the AFCS differential authority requirements in the DELS. The pilot can override the CCDA at any time.

Control force ard control position trim functions are provided. Longitudinal, lateral and directional forces can be trimmed to zero by the cyclic grip trim switch (equivalent to conventional magnetic brake switch). A force trim switch is also provided on the collective lever trip. If the pilot is holding a force and wishes to retrim, he operates the switch which opens the magnetic brake. The feel spring is free to collapse. The retrim damper smooths the motion.

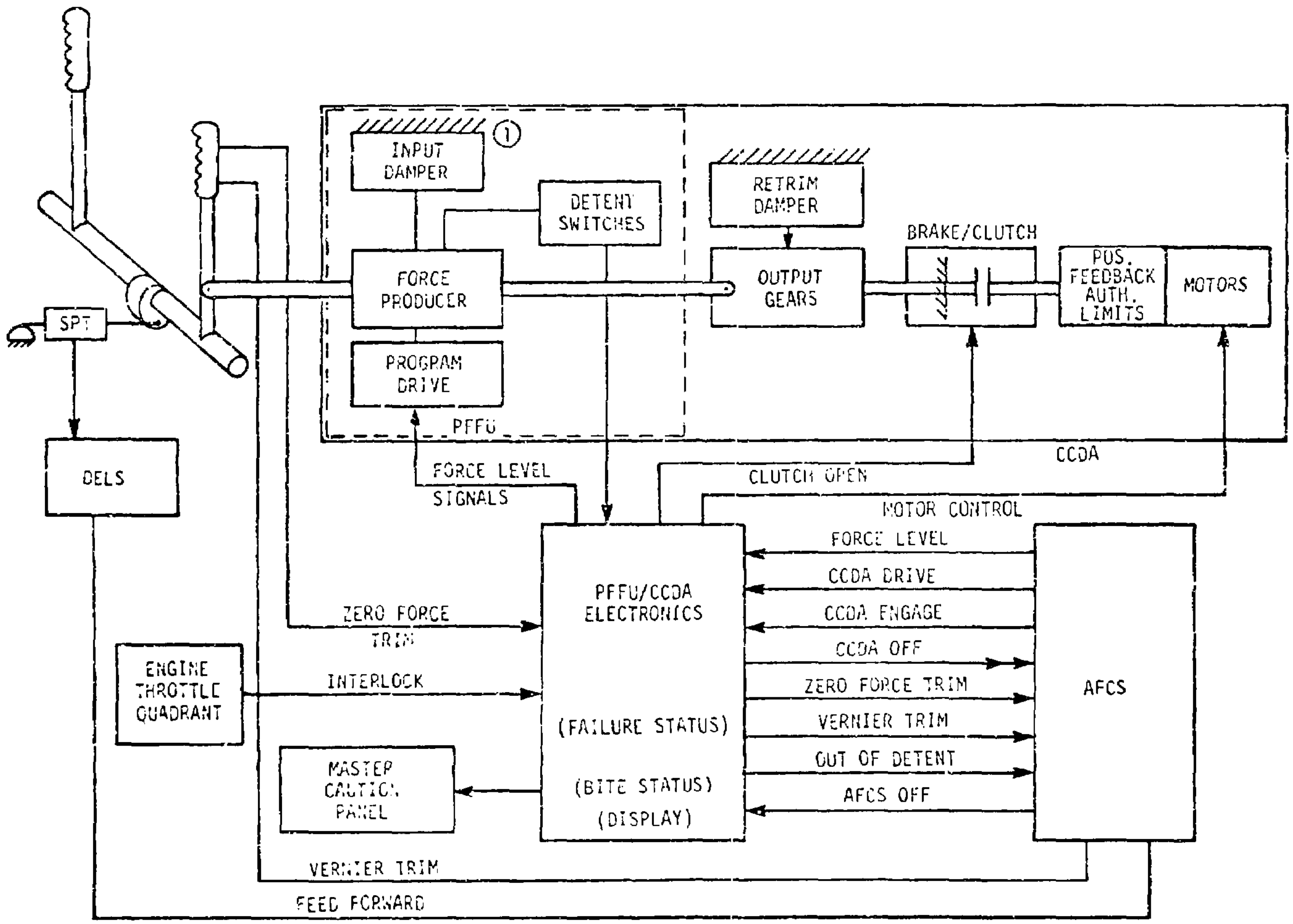
PFFU/CCDA ATC Hardware

A photograph of a ship set of PFFU/CCDA equipment is shown in Figure 47. There are four actuator units. Two identical electronics units (dual redundancy) complete the hardware.

PFFU/CCDA Redundancy

The electronic units are completely dualized to provide a fail-operate system. An "active standby and model" method of redundancy is provided. The system will continue to operate normally after one failure. If a second failure of the same kind occurs, the monitor switches control to the active standby channel. The input signals from the AFCS are monitored so that a failure in one AFCS output will cause switch-over to the active standby channel.

The electro-magnetic parts of the actuators are dually redundant. The following items are dualized:



NOTE: (1) DAMPERS GROUNDED TO CASE

Figure 46. PFFU/CCDA Block Diagram.

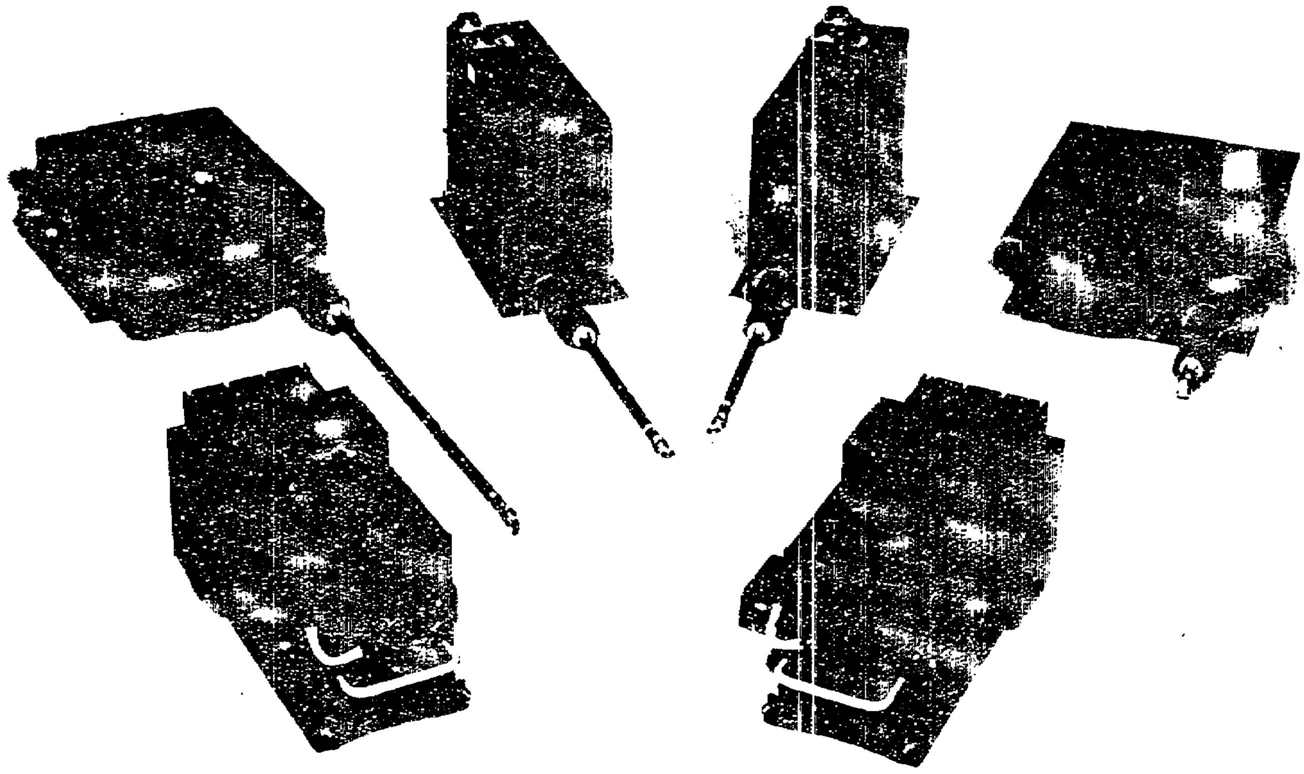


Figure 47. Programmable Force-Feel Unit/Cockpit Control Driver Actuator System (PFFU/CCDA).

1. PFFU and CCDA Servo Motors.

2. PFFU and CCDA Clutches.

3. PFFU and CCDA Position Transducers.

4. PFFU and CCDA Magnetic Brake Coils.

Built-In-Test Equipment (BITE)

BITE is incorporated together with a self-contained display and control panel on the front of the electronics units. The BITE function is interlocked with the engine condition levers in such a way that the BITE cannot be run while the engines are operating. A comprehensive set of tests are run and all the circuits are exercised in an automatic sequence. If there is a failure, the test sequence stops and the test number is displayed by the indicating lights on the BITE panel.

TEST RESULTS

Laboratory tests of the CCS used the test stand pictured in Figure 37. Testing was divided into two major parts: the mechanical system with PFFU/CCDA disconnected and the complete system with PFFU/CCDA connected. Results are reported in the following paragraphs.

Mechanical

The mechanical system was tested with the PFFU/CCDAs disconnected. The following results were obtained.

Interference Check

The controls were moved singularly and simultaneously to search out interferences. Some interference was found and corrected. No interference was of sufficient magnitude to discredit the design approach.

Friction and Looseness Checks

Initially, the mechanical controls operated in a generally satisfactory way, but three kinds of defects were found. The defects were: (1) backlash in shear-pin joints and other pin joints, (2) tightness, or high friction, in the ball bearings,

and, (3) high drag caused by excessive flexing of the stick position transducer wire bundles and other control stick control wires. There were many joints that had zero backlash, some bearings that had very low friction and many wire bundles that presented little or no drag on the controls. It was concluded therefore that, with improved requirements for workmanship and inspection, and with only very minor design changes in the wire bundle arrangement, that the performance would be quite satisfactory. The test stand was reworked in order to proceed with testing.

The final data on friction and looseness is shown in Table 5. The directional breakout and running friction is considered to be too high. A drawing change was made to use self-aligning bearings for torque tube mounting.

Shear Out Checks

One shear out was performed on each axis. The force required to shear and the friction after shear was noted. The test values are tabulated below. The results were considered satisfactory. A jury bar was installed to represent a jammed stick position transducer (SPT) or PFFU/CCDA.

Longitudinal SPT:	30 lbs. to shear 1.0 lbs. drag
Lateral SPT:	38 lbs. to shear 1.8 lbs. drag
Directional SPT:	80 lbs. to shear Negligible increase in friction.
Collective SPT:	48 lbs. to shear 2.0 lbs. drag
Longitudinal PFFU/CCDA:	70 lbs. to shear 0.4 lbs. drag
Lateral PFFU/CCDA:	40 lbs. to shear 1.5 lbs. drag
Directional PFFU/CCDA:	90 lbs. to shear 6 lbs. drag
Collective PFFU/CCDA:	45 lbs. to shear 6 lbs. drag

TABLE 5. FINAL DATA ON FRICTION AND LOOSENESS

CONTROLLER AXIS	WORSE CASE VALUES WITH SPTs CONNECTED		
	BREAKOUT FORCE-LBS.	RUNNING FORCE-LBS.	LOOSENESS
Longitudinal	0.8	0.7	None Perceptable
Lateral	0.55	0.4	None Perceptable
Directional	5.5	--	Perceptable, but insignificant.
Collective	1.8	1.8	None Perceptable

SPT Tracking

After the rigging checks and the stops were set, the tracking of the SPTs (linear voltage differential transformers) was measured on the longitudinal and lateral axes. The maximum spread between the six SPTs per axis is plotted in Figure 48. Tracking was within tolerance.

Complete System Checks

The PFFU/CCDAs were connected to each axis, and a series of tests to acquire functional data were made. The significant and design substantiating data are given below.

Variable Force Feel

Forces at the grips (and pedals) were measured as the input signals to the PFFU were varied. The measured data is shown in Figures 49 through 52. It was concluded that the performance was satisfactory.

Each level of force feel produces a different breakout force and different gradient. These two parameters provide the cues to the pilot when the AFCS is programmed to drive the PFFU.

CCDA Data

Figure 53 shows typical data taken to measure the drive rate capability of the CCDA. Tests were made for various values of force feel. The overall results for zero volts and ten volts PFFU signal are tabulated in Table 6. The CCDA driving function is considered satisfactory.

CCDA, Stick Position versus Command Signal

Command signals to the CCDA were simulated, and the resulting controller positions were measured. Typical test data is shown in Figures 54 and 55. The test data shows that the CCDA performance was satisfactory in each control axis.

Input Damping

The terminology refers to damping of controller upon release by the pilot after a displacement. Typical test data is shown in Figures 56 and 57. Table 7 presents the reduced input damping data for several of the PFFU control voltages.

The damping performance was generally satisfactory, but in a few cases, it seemed somewhat low. However, since neither inertias of the actual system nor the preferences of the pilot were known exactly, the results were considered acceptable. It was observed that the dampers were set at the

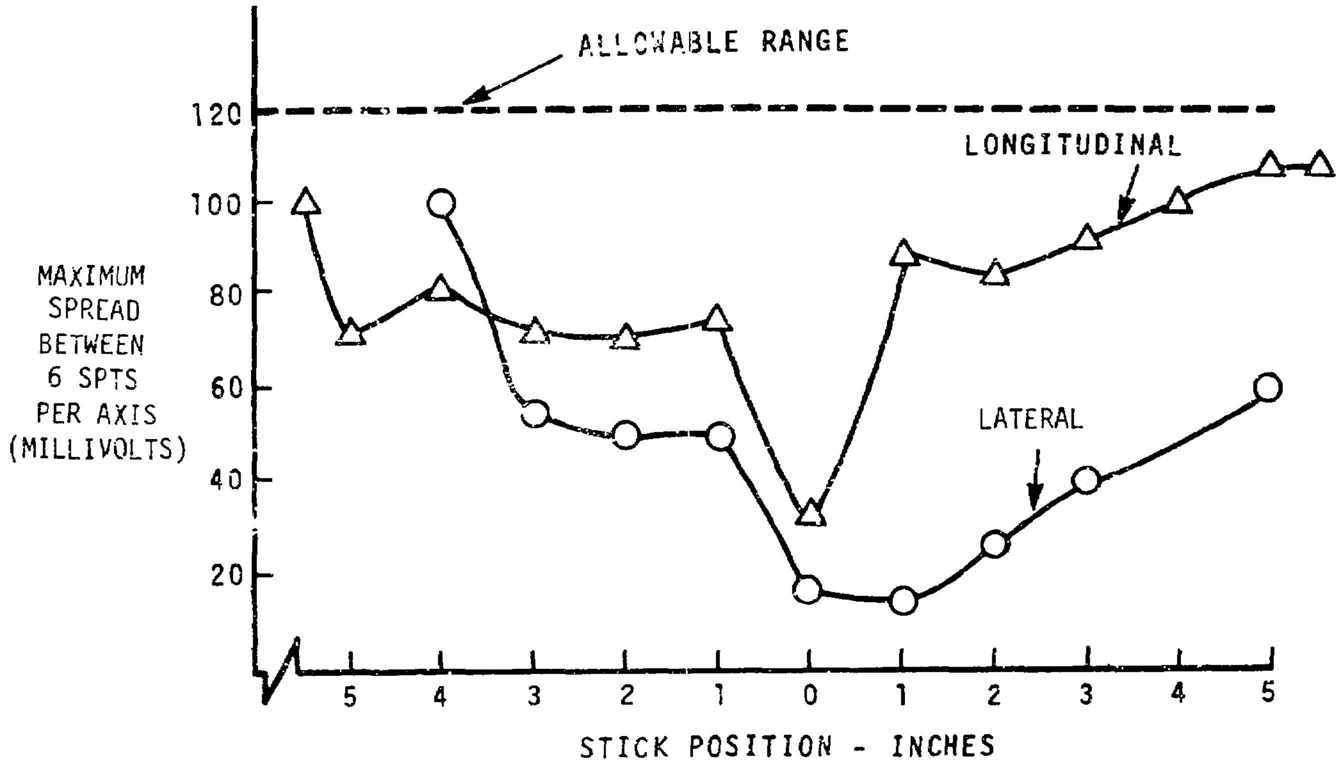


Figure 48. SPT Tracking Data.

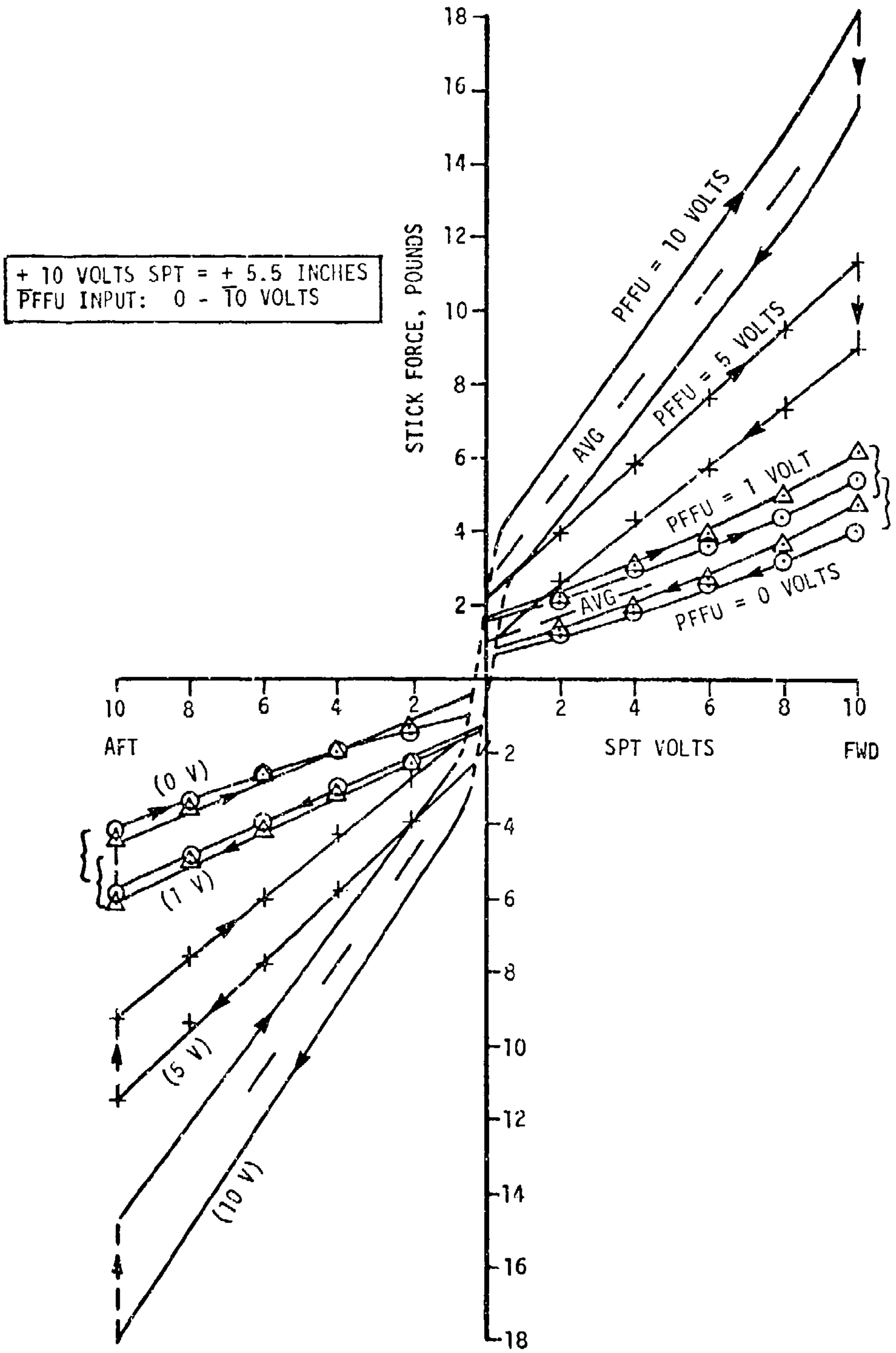


Figure 49. Longitudinal Variable Force Feel.

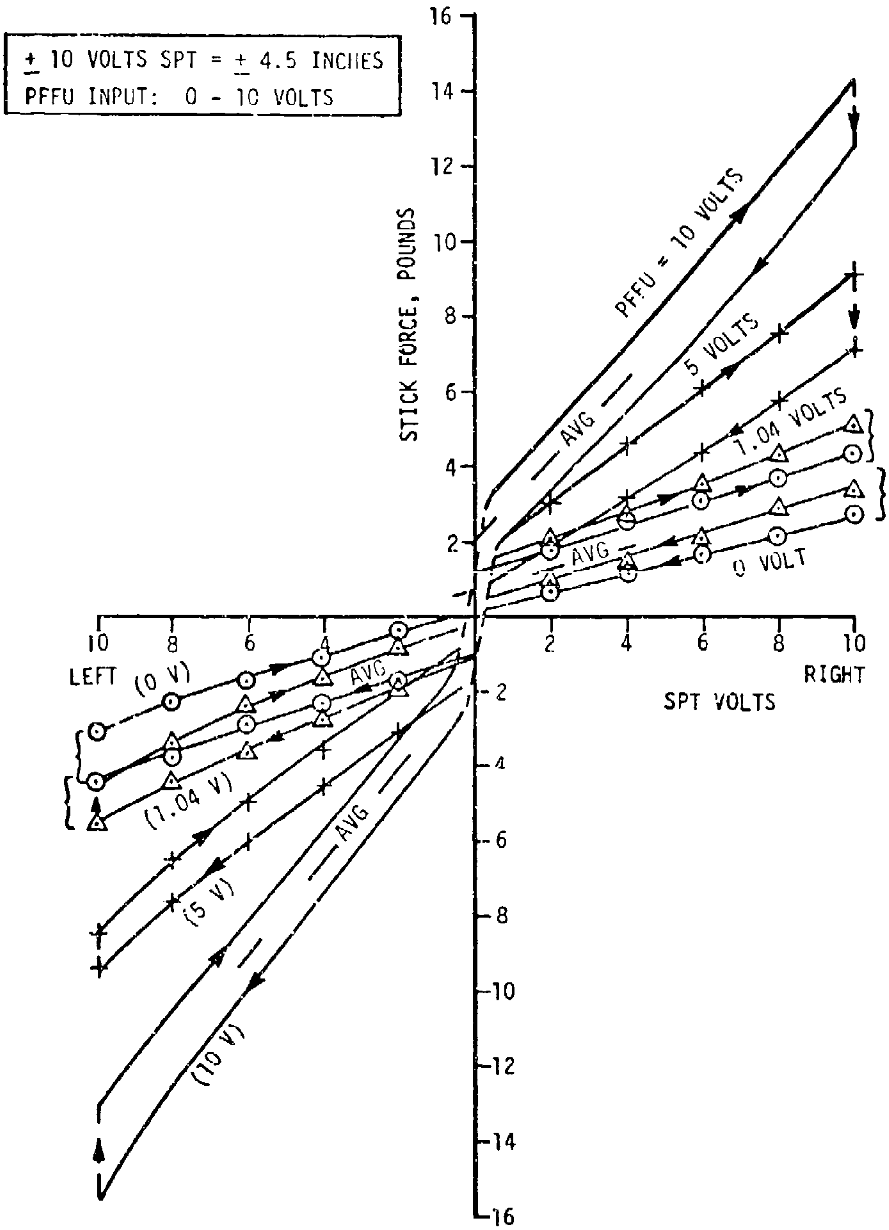


Figure 50. Lateral Axis Variable Force Feel.

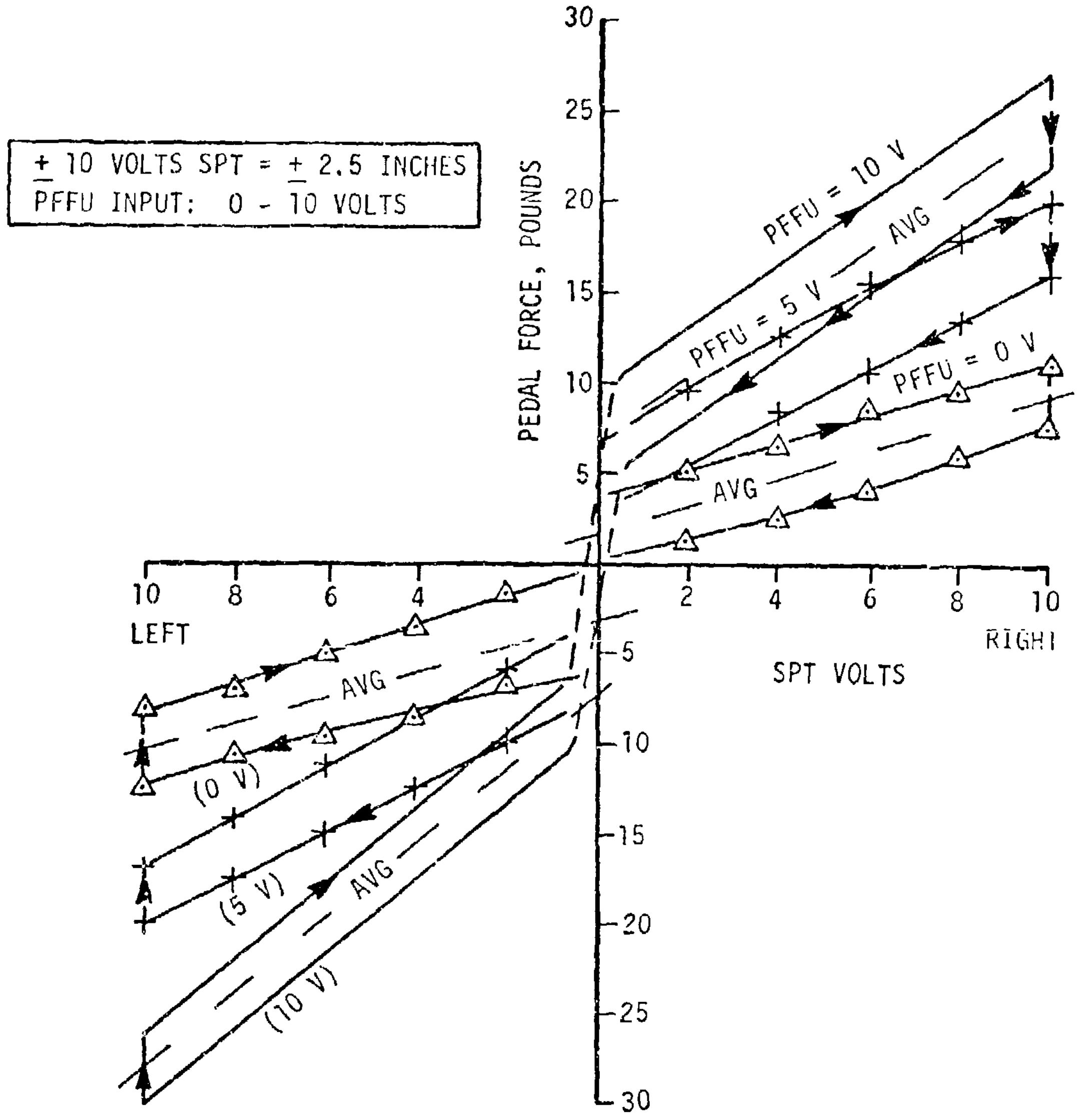


Figure 51. Directional Axis Variable Force Feel.

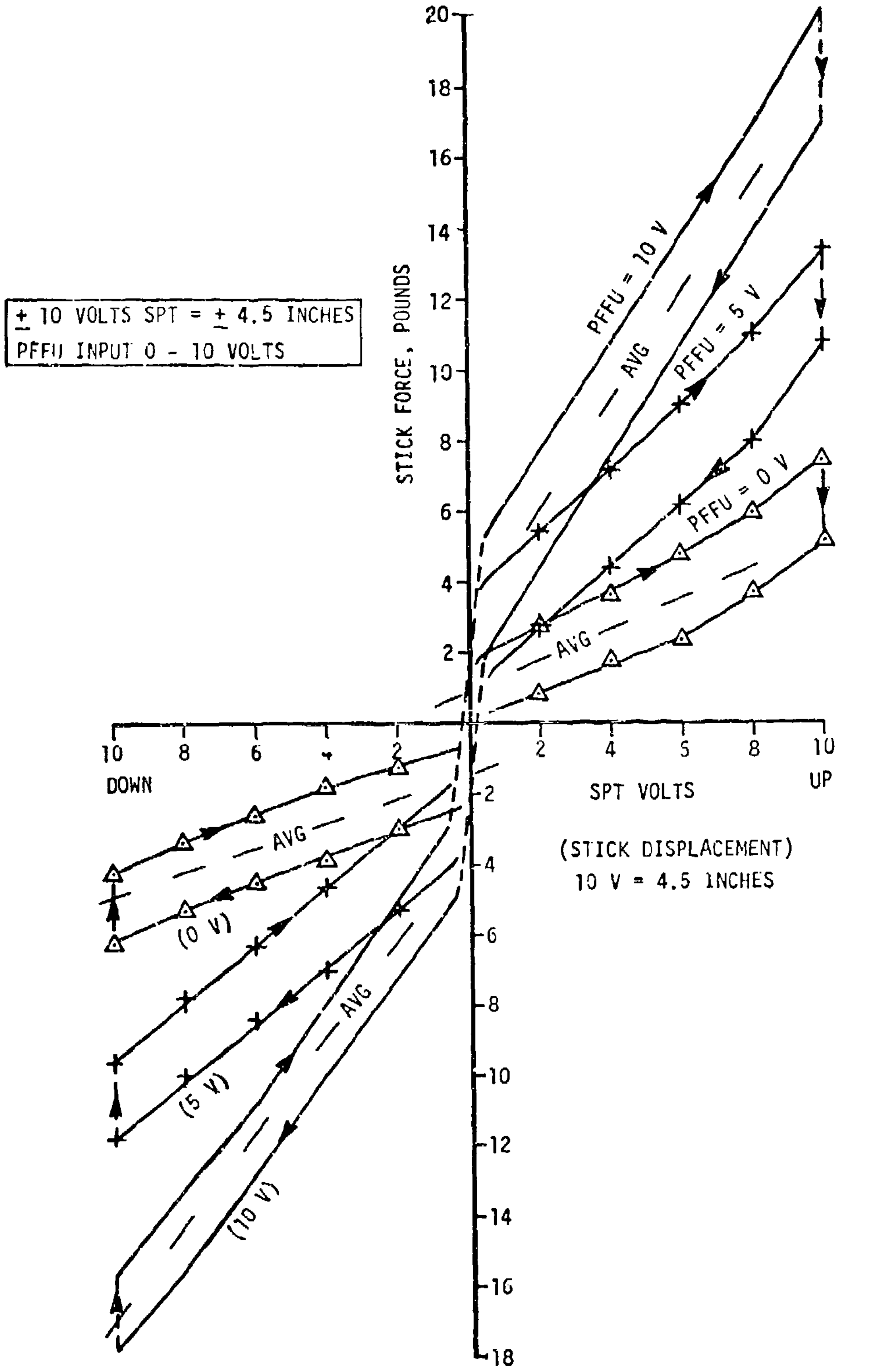


Figure 52. Collective Axis Variable Force Feel.

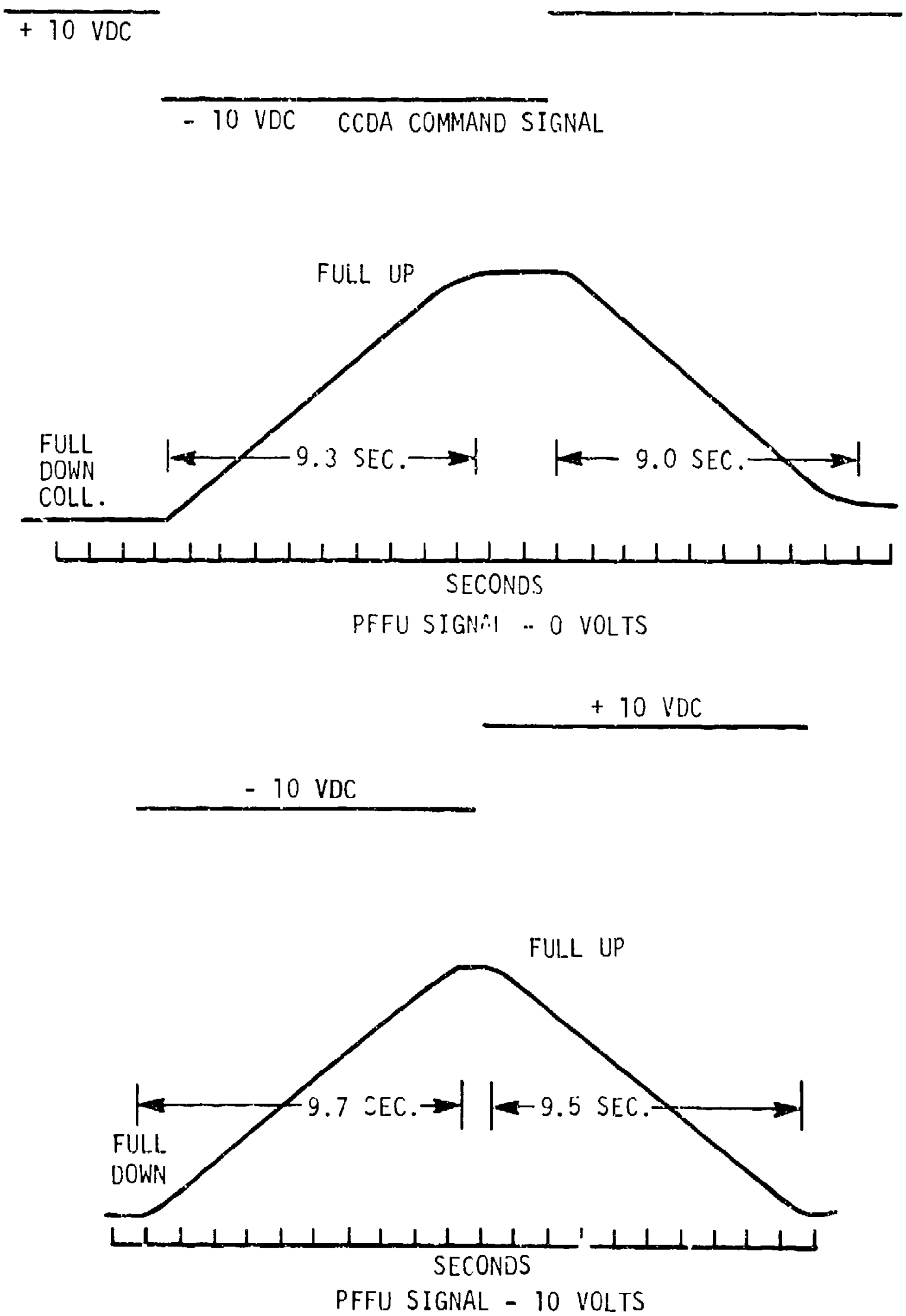


Figure 53. Typical Data, CCDA Drive.

TABLE 6. CCDA DRIVE RATES AT TWO LEVELS OF FORCE FEEL

AXIS	DIRECTION	RATES - IN/SEC		GOAL
		0 VOLTS PFFU	10 VOLTS PFFU	
Longitudinal	Fwd	1.1	1.2	1.08 - 1.12
	Aft	1.1	1.15	
Lateral	Left	1.14	0.82	0.8 - 0.9
	Right	1.3	0.83	
Directional	Left	0.52	0.51	0.5 - 0.55
	Right	0.52	0.53	
Vertical	Up	0.98	0.93	0.9 - 1.0
	Down	1.0	0.95	

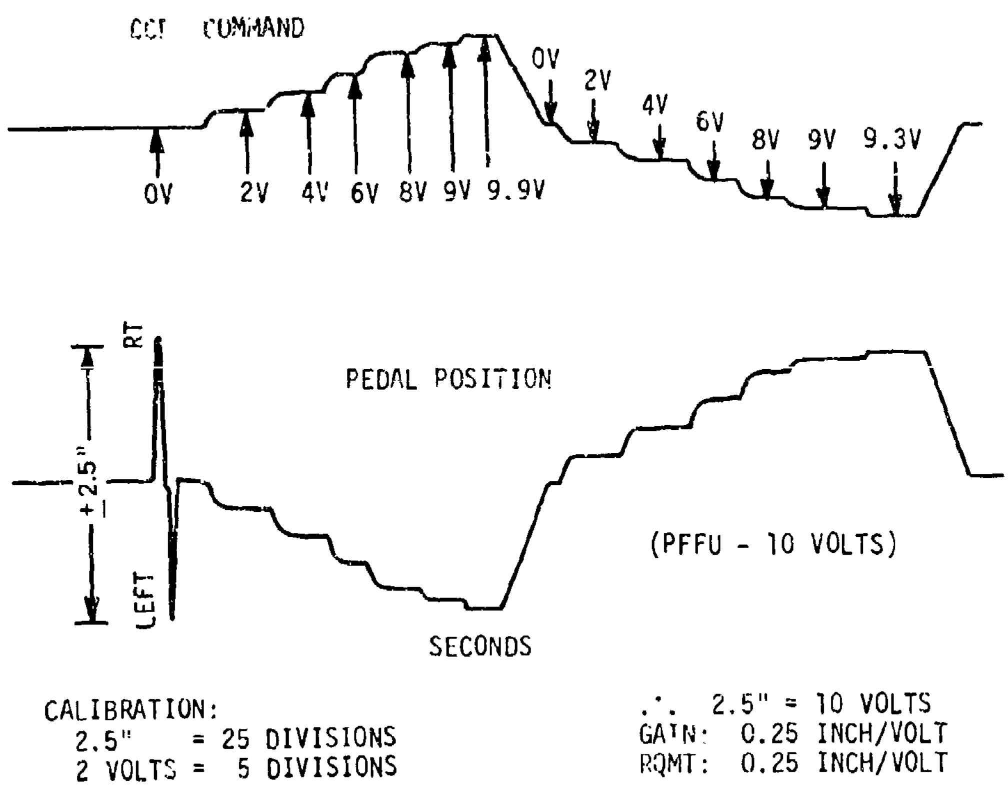


Figure 54. Typical Data, Pedal Position Versus CCDA Command Signal.

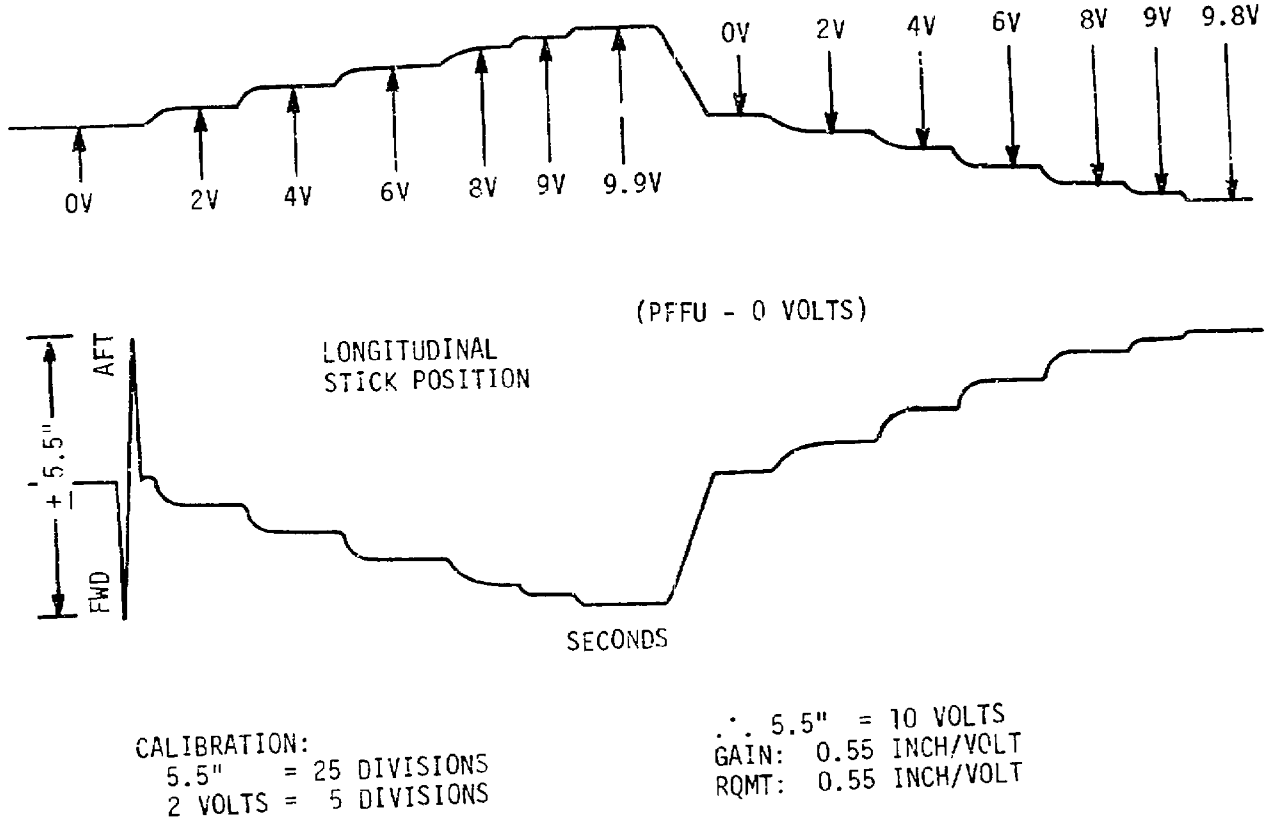
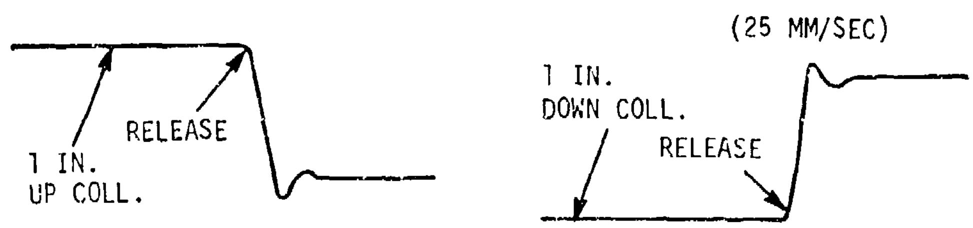
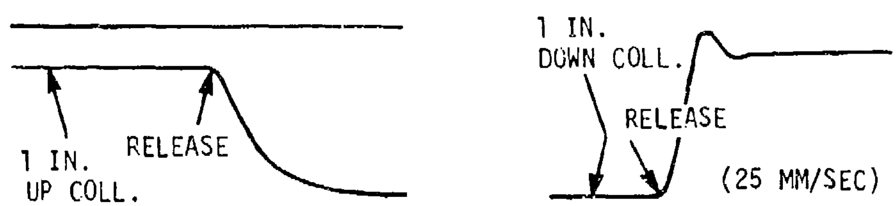


Figure 55. Typical Data, Stick Position Versus CCDA Command Signal.



PFFU VOLTAGE - 10 VOLTS



PFFU VOLTAGE - 0 VOLTS

Figure 56. Typical Data, Input Damping, Collective Axis.

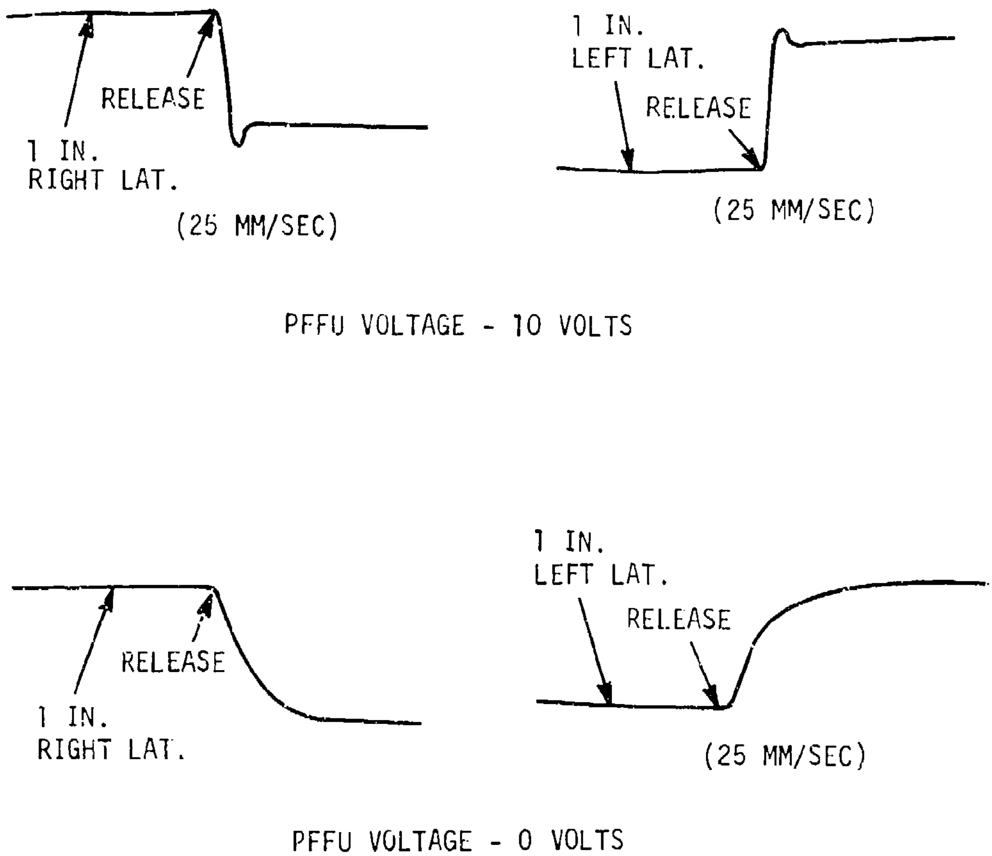


Figure 57. Typical Data, Input Damping, Lateral Axis.

TABLE 7. TABULATION OF INPUT DAMPING TEST DATA

AXIS PFFU CONTROL VOLTS	FIRST OVERSHOOT	NO. OF OVERSHOOTS	DAMPING RATIO
Longitudinal			
0.0 Volts	20 & 28%	1 & 2	.6 & .5
1.0 Volts	24%	2	.5
5.0 Volts	26%	3	.4
10.0 Volts	32%	3	.4
Lateral			
0.0 Volts	None	None	1.0
1.04 Volts	None	None	1.0
5.0 Volts	6%	1	.7
10.0 Volts	15%	2	.5
Directional			
0.0 Volts	None & 2%	None & 1	1.0 & .8
5.0 Volts	12%	2	.65
5.8 Volts	14%	2	.5
10.0 Volts	15%	2	.5
Collective			
0.0 Volts	None & 16%	None & 2	1.0 & .5
4.35 Volts	14% (12%)	2	.5
5.0 Volts	16% (12%)	2	.5
10.0 Volts	12% (12%)	2	.55

highest ends of their adjustment ranges, and it was decided to call for a heavier fluid for some components. The programmable feature of the damping proved to be quite satisfactory in producing a nearly constant damping ratio over the programmable range of force levels. Overall, the design was quite satisfactory.

Force Retrim Damping

The terminology refers to the controller damping when the "magnetic brake" is used to trim the stick forces to zero, at which time the programmable force-feel spring collapses to neutral. No strip chart data was taken. However, pilot comment indicated that the damping was satisfactory in longitudinal, excessive in lateral, and under damped in collective and directional. The dampers are adjustable and can be reset. They were not reset during the tests.

Hysteresis and Backlash at Detent

The backlash requirement placed on actuators was 0.9% half-travel or .0135 inch at the actuator. Results measured on the actuators were as follows:

Longitudinal	.007 inch[1]	Directional	.003 inch[2]
Lateral	.004 inch[1]	Vertical	.010 inch[2]

Note

Number one taken at low force gradient position of PFFU, and number two taken at the high force gradient position of PFFU in order to be representative of the prototype values.

The mechanical controls evidently are capable of extremely low backlashes, perhaps close to zero, but a tentative requirement decided on was 0.3 percent. Using this value, the system requirements would be 1.2 percent.

AXIS	BACKLASH PFFU-MEASURED (Inch)	PFFU RQMT. (Inch)	PROPOSED MECH. CONTROLS RQMT. (Inch)	SYSTEM RQMT. (Inch)
Longitudinal	.026 @ Stick	.050	.016	.066
Lateral	.011 @ Stick	.036	.012	.048
Directional	.005 @ Pedals	.022	.008	.030
Vertical	.030 @ Stick	.040	.014	.054

The hysteresis requirement placed on the actuators was seven percent of half-travel away from detent. Measurements were made on two actuators at the supplier's plant as representative of the log of four. Results were as follows:

AXIS	HYSTERESIS REQUIREMENT (Inch)	MEASUREMENTS AT DETENT (Inch)	MEASUREMENTS OUT OF DETENT (Inch)
Longitudinal	.385	.100	.45
Lateral	.280	.040	.20

These results were considered to be satisfactory, especially in light of the fact that vibration of the aircraft will break up stick friction and thus reduce the hysteresis.

Hysteresis measurements made on the complete system yielded the results shown in Figures 58 and 59 for expanded views of longitudinal and lateral axes. (No data was available from the supplier for the directional or vertical axes, so no system measurements were made for these two axes.) The hysteresis results for all four axes are portrayed in Figures 49, 50, 51, and 52. The hysteresis at detent was considered to be satisfactory under the condition that the sticky friction of the mechanical controls will be reduced to nearly zero. (This condition was simulated by a low-frequency vibration of 2 Hz applied to the structure to break up the stick friction.) Note that the hysteresis measured was approximately four percent in all axes. For follow-on equipment it should be approximately two percent.

Detent Switches

The switches did not meet their requirements either in tracking or in points of actuation. The cause was found to be inherent in the design. Backlash and flexing of other parts caused the actuation points to be located further away from the neutral position than was required. Tolerances on the mechanical parts associated with the switch package and the difficulty in making adjustments caused the failure to meet tracking requirements.

The original specifications were as follows:

1. Actuation points adjustable between 1-4 percent.

2. Tracking accuracy, 0.5 percent.

After an analysis of the design, it was decided that the best results which could be obtained were as follows:

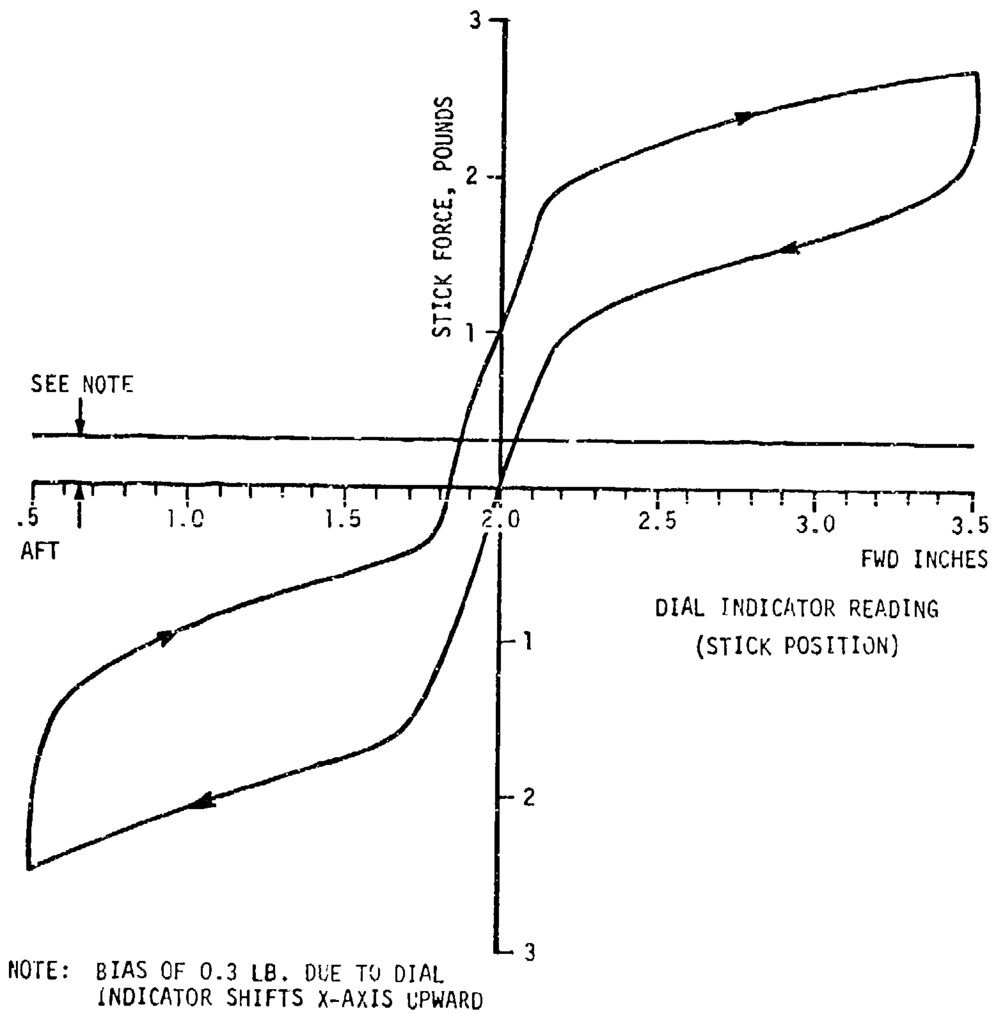


Figure 58. Longitudinal Axis Hysteresis at Detent, Complete System.

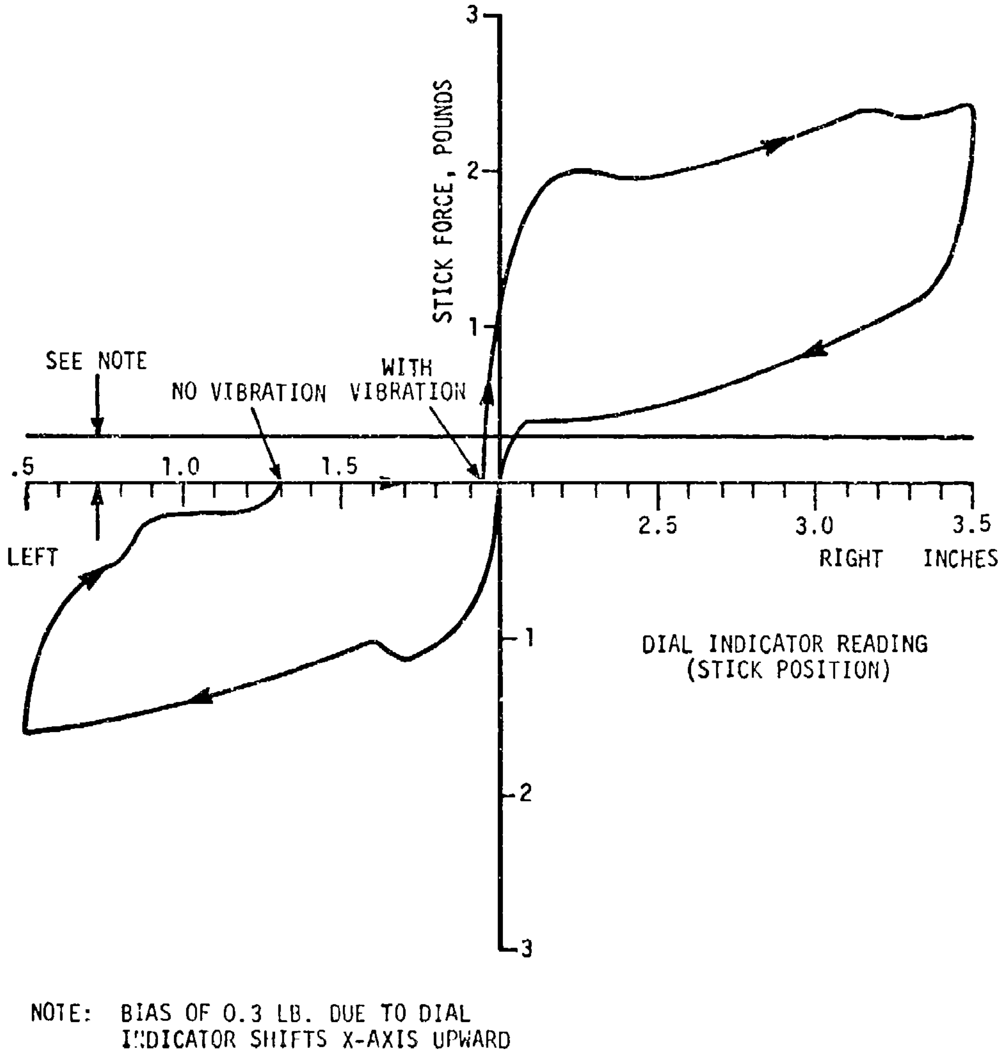


Figure 59. Lateral Axis Hysteresis at Detent, Complete System.

1. Actuation points set between 3 and 4 percent.

2. Tracking accuracy, 1 percent.

BITE Evaluation

The built-in test equipment was found to be satisfactory. A series of faults were introduced, and BITE functioned perfectly in each case.

Pilot Evaluation

The initial evaluation of the test stand revealed some play in the shear devices and some friction due to slight misalignment of bearings. These problems were corrected by mount adjustment and bushing the shear pins. Several torque tube support bearings were replaced with self-aligning bearings to ensure alignment with minimum mount adjustment. Following these minor modifications, the mechanical controls were judged to be free of detectable looseness, and friction levels were insignificant.

Control throws and balance were good. Pedal adjustment was good; however, the curved heel slide was unacceptable because the minimal pedal-to-rest clearance and the steep slope of the rest curve results in interference with the shoe heel. This problem will be corrected in the HLH prototype.

The PFFUs in all axes were evaluated, and force levels throughout the programmable range were acceptable. PFFU response frequency tests were conducted to determine that force programming rates were adequately high to enable programming during pilot control displacements with no apparent lag in force buildup.

These tests were conducted by programming force changes during pilot stop control displacements. PFFU response was adequate to program force changes very rapidly with no apparent lag in force buildup, thus demonstrating system's ability to provide adequate force change rates to accomplish all conceivable force feel cueing requirements, including envelope limiting.

Shear device operation was evaluated by simulating jams, and the shear forces were acceptable. No increase in system friction was evident following shear. Due to control mass balancing, all axes except collective pitch were judged acceptable following a shear of the PFFU/CCDA input. The collective lever is completely free of friction and damping and may contribute to pilot-induced oscillation (P.I.O.). Additional control viscous damping is required to damp the control to the structure. This additional damper will improve damping for normal PFFU operation and will ensure control stability following a shear of the PFFU/CCDA input.

CONCLUSIONS - COCKPIT CONTROLS SUBSYSTEM

The basic design approach of the mechanical elements was found to be satisfactory for refinement and implementation in the HLH. Care must be taken to minimize any looseness that can be caused by shear pins. Friction must be minimized in roller bearings. Suggested approach is to use self-aligning bearings.

The bench test shows that a damping device should be incorporated between the collective lever and the structure to keep the lever from being completely free upon shear of the PFFU/CCDA connecting shaft.

Static evaluation of PFFU/CCDA shows that the design concept can provide the desired functions. The static testing does not confirm that there is a need for programmable force feel in the HLH.

RECOMMENDATIONS - COCKPIT CONTROLS SUBSYSTEM

It is recommended that the basic design and function of the mechanical elements be utilized in the HLH production program. Attention to design detail and selection of roller bearings is required to minimize friction. A damping device between the collective lever and the structure should be considered.

A requirement for programmable force feel could not be determined in static test, and the ATC test program shows that the HLH will be a very stable and controllable aircraft. It is recommended that programmable force feel should not be incorporated in the production HLH program. The requirement for such function should be based on flight evaluation of a prototype vehicle.

LIST OF SYMBOLS

AFCS	Automatic Flight Control System
APU	Auxiliary Power Unit
ATC	Advanced Technology Component
BITE	Built-In-Test Equipment
CCDA	Cockpit Controller Driver Actuator
CCS	Cockpit Controller Subsystem
EMC	Electromagnetic Compatibility
EHV	Electro-hydraulic Valve
FBW	Fly-By-Wire
HLH	Heavy Lift Helicopter
Hz	Cycles per second
K	Knots
KIAS	Knots Indicated Airspeed
LCP	Longitudinal Cyclic Pitch
LED	Light Emitting Diode
LVDT	Linear Voltage Differential Transformer
MTBF	Mean Time Between Failure
PFCS	Primary Flight Control System
PFFU	Programmable Force Feel Unit
N_R	Rotor Speed, Revolutions per Minute
psi	pounds per square inch
RFI	Radio Frequency Interference
SCAS	Stability and Control Augmentation System
SDA	Swashplate Driver Actuator
SPT	Stick Position Transducer

UBA	Upper Boost Actuator
VAC	Volts, Alternating Current
VDC	Volts, Direct Current
$\Delta\rho$	Differential Pressure

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2. HLH/ATC Flight Controls DEL Ground and Flight Test Report, HLH/ATC Task II, dated December 19, 1973